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Calspan

Technical Report

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Calspan

On November 17, 1972 Cornell Aeronautical Laboratory (CAL) changed its name to Calspan Corporation and converted to for-profit operations. Calspan is dedicated to carrying on CAL's long-standing tradition of advanced research and development from an independent viewpoint. All of CAL's diverse scientific and engineering programs for government and industry are being continued in the aerosciences, electronics and avionics, computer sciences, transportation and vehicle research, and the environmental sciences. Calspan is composed of the same staff, management, and facilities as CAL, which operated since 1946 under federal income tax exemption.

PLUME INTERFERENCE ASSESSMENT AND MITIGATION

LABORATORY MEASUREMENTS PROGRAM

Calspan Report No. KC-5134-A-6

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FOREWORD

A research program directed toward investigating possible exhaust plume radiance/IR sensor interaction is currently underway. The program is sponsored by the Army Ballistic Missile Defense Agency, with Mr. T. Roy Bowen of ABMDA, Huntsville, Alabama, the program technical monitor. Calspan Corporation, formerly Cornell Aeronautical Laboratory, Inc., of Buffalo, New York, is the prime contractor for the overall program, which consists of three main tasks. The initial research efforts for two of the tasks have been subcontracted by Calspan to the AVCO Everett Research Laboratory (AERL), of Everett, Mass., and Aerodyne Research, Inc. (ARI) of Burlington, Mass.

The overall requirement of the research program is to develop a comprehensive understanding and characterization of high altitude plumes from interceptor rocket motors which is adequate for the evaluation of the effects of the plume on the signal-to-noise ratio of IR sensors, and the identification of the parameters which contribute most strongly to the problem. To satisfy this requirement, the program has been structured into three main tasks:

I. Laboratory measurements

II. Near-field analysis and prediction

III. Far-field analysis and prediction.

Calspan Corporation, the prime contractor for the overall program, is responsible for the experimental measurements under Task I. Under separate subcontracts, the AVCO Everett Research Laboratory performed near-field analyses under Task II, while the far-field analysis was conducted by Aerodyne Research, Inc. for Task III.

The final report for the overall program is being distributed in several volumes, consistent with the task structure noted above. Accordingly, the far-field effort is discussed in a 3-volume report by ARI, and the near-field analysis is reported in a single AERL volume. The present report discusses the laboratory measurement program performed at Calspan.

An overall summary of each volume, a discussion of the program structure, and a bibliography of pertinent reports published under this contract are present in a separate summary report entitled:

Plume Interference Assessment and Mitigation; Final Report. Calspan Report No KC-5134-A-7, June 1973. Submitted by P.V. Marrone.

ACKNOWLEDGMENTS

Several members of the Aerodynamic Research Department at Calspan are principal investigators in the experimental program. These include Mr. K.C. Hendershot and Mr. T.M. Albrechcinski, who are responsible for rocket motor design and performance; Mr. R.A. Fluegge, who directs the particle collection and analysis phase of the program; Dr. W.H. Wurster and Mr. J.E. Stratton, who are responsible for the optical diagnostics and radiation signature measurements; and Dr. C.E. Treanor, who performed analyses in specific problem areas of exhaust plume radiance characterization.

SUMMARY

A rocket plume diagnostic program is currently underway at Calspan. The primary purpose of the program is to investigate the high-altitude near-field radiance signature of rocket motors, and to develop the diagnostic techniques required to characterize various types of rocket plumes. The design, calibration, and installation of a cold-optics LWIR detecting system on one of the Calspan high-altitude test chambers is discussed. Particulate flux measurement techniques are discussed, and the design and development of research rocket motors for use with specific propellant combinations is presented. A test program for small liquid and solid thrusters, such as those proposed for the FAIR II PIE flight program, has been initiated and experimental data for a monopropellant hydrazine thruster are presented and compared with IR radiance predictions.

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Section 1
INTRODUCTION

Missile-borne IR sensor performance may be degraded due to the presence of large exhaust plumes from any nearby rocket motor. Spurious radiation signals may arise either from the self-luminous near-field of these exhaust plumes, or from the direct excitation of highly cooled exhaust products upon interaction with the tenuous atmosphere at distances far from the exhaust nozzle. In addition, the sensor performance may be degraded due to the impingement or deposition of particles and gases on the cold surfaces of the sensor system. The magnitude of the reduction in sensor signal-to-noise ratio is dependent on several factors, including: the altitude and velocity of the vehicle; the rocket fuel/oxidizer combination; the angle between the plume axis and vehicle velocity vector; and the optical sensor field of view.

Under a previous program, which was explicitly directed to the special case of the HIT vehicle, many of the techniques required for the characterization of rocket motor plumes were developed.(1) For example, gasdynamic and optical techniques for laboratory measurement of near-field plume properties were demonstrated.

The scope of the present laboratory measurements program is to provide the Army Ballistic Missile Defense Agency (ABMDA) with experimental studies which will characterize plumes from rocket motors at high altitudes, including both main boosters and ACS motors. Various fuel/oxidizer combinations and nozzle configurations will be under evaluation for compatibility with far IR sensors. The experimental techniques previously developed have been modified and extended during the course of the present program. For example, the IR spectral range previously investigated has been extended to better cover the wavelength region of interest to midcourse optical systems.

Section 2
LABORATORY MEASUREMENTS PROGRAM

Under the Laboratory Measurements phase of the overall Plume Interference Assessment and Mitigation Program, there are two main tasks. The first is a series of rocket propellant screening experiments under which the near-field far IR plume radiance characteristics will be investigated for several liquid and solid propellants, nozzle types and motor configurations. The second task is to obtain laboratory measurements of the near-field radiation from plumes of small thrusters of the type to be flown on the FAIR II plume interaction flight experiments. To accomplish these two main tasks, several subtasks are necessary and will be discussed in this section of the report. These subtasks include the design, fabrication, calibration, and installation of the cooled far-IR optical diagnostic system, the fabrication of additional particle collectors, and the design of several of the rocket motor configurations to be studied during the test program.

The experimental rocket firings are conducted in one of the large, high-altitude test chambers at Calspan. The 10-foot-diameter by 28-foot-length chamber is capable of being evacuated to pressures on the order of 5×10^{-5} torr by means of a large refrigerated-chevron-baffled oil diffusion pump.

The available test time in the altitude chamber is determined by the time required for the plume gas to reach the end of the chamber and propagate a disturbance back to the motor station (typically 15 ms).[2-5] That is, until the fact that the test chamber is of finite size is communicated back to the test region (by gas dynamic wave processes), the exhaust plume flow field behaves as if it were operating in an infinitely large test volume. Since the actual test altitude is that corresponding to the ambient pressure in the vacuum chamber immediately preceding rocket firing, continuous pumping of the rocket exhaust products is not necessary to maintain test altitude during the short test time; thus, a vacuum pumping capability adequate to bring the tank to the desired test altitude in a reasonable time is all that is required.

2.1 OPTICAL DIAGNOSTIC DEVELOPMENT

The extensive current interest in plume physics and radiation spans a broad spectrum of wavelengths. Wavelengths as short as 1000 Å in the vacuum ultraviolet may be relevant to rocket launch phenomena, while radiation in the far infrared is of interest to interceptor technology. This broad span of wavelengths, some of the radiating species and pertinent transitions involved are diagrammatically shown in Figure 1.

The wavelength range between 0.1 and 40 μm has been simply divided into three regimes, which, by virtue of the different excitation energies involved, correspond to three basically different types of emission spectra from the various species shown. In general, radiation between 0.1 and 1.0 μm involves transitions between electronic states of the molecules. There results a broad band system, rich in lines from the many vibrational and rotational states populated at the equilibrium temperature necessary to induce measurable emission. However, this is not the case for rocket motor exit plane temperatures below 1000°K, because equilibrium excitation energies are too low. Thus, as is the case for CO 4+ band emission around 1500 Å, chemiluminescent excitation mechanisms are usually postulated to populate the upper states involved.

The region between 1.0 and 10 μm is of special interest to plume diagnostics. Most species in the plume which are infrared active (permanent dipole moments) undergo vibration-rotation transitions which require modest excitation energies, and which permit their presence, their concentrations and their degree of excitation to be assessed by emission and absorption measurements. For this reason, because the main thrust of the programs at Calspan are primarily diagnostic in nature, this spectral region is measured quite thoroughly. A recently completed solid propellant plume testing program for ARPA involved spectral scans in the wavelength region between 2 and 8 μm.[2] It is specifically planned to utilize the SWIR 2-8 μm rapid scan spectrometer during the ABMDA test program. Finally, because the far IR is of special and direct concern to ABMDA objectives, spectral scanning instruments between 6-12 and 12-24 μm are being implemented for this program.

Each of the principal optical diagnostics will be separately discussed below. They are shown schematically deployed in Figure 2.

2.1.1 SWIR Rapid-Scan Spectrometer

For SWIR measurements under an ARPA-sponsored plume diagnostic program,[2] a scanning spectrometer was designed and fabricated which employs a circular variable filter wheel rotating in front of two detectors. Since this instrument will be utilized during a part of the ABMDA experimental test program, it will be briefly described herein.

A two-element detector array is focused onto the rocket plume axis by a silicon lens, see Figure 3. In the optical train near the detectors is mounted a circular variable filter (CVF) obtained from Optical Coating Laboratories, Inc., Santa Rosa, California, which is spun at 8000 rpm by a 400 Hz synchronous motor. The CVF is made in two semicircular segments spanning the wavelength ranges 2-4 and 4-8 μm. One detector is InSb, covering the range between 2 and 5.5 μm. The balance of the spectrum is recorded by the second detector, which is Ge:Au. This arrangement thus permits the entire spectrum to be scanned in 7.5 ms. Since nominal rocket test times are between 15-30 ms, this readily permits several spectra to be recorded during this interval. The wavelength resolution is determined by the filter characteristics and the azimuthal angle of the CVF subtended by the detectors. The CVF has a 2% half-bandwidth, which corresponds to a resolution between 0.04 and 0.16 μm. This resolution enables species identification to be made.

The detector areas are each 1 x 3 mm, and with a magnification of approximately 3, the plume area subtended by each is 3 x 9 mm. The use of two detectors results in the sampling of two adjacent areas. The wavelength overlap permits the two channels to be correlated.

Calibration of the system is achieved by direct comparison with the radiation from a standard black body source situated within the high altitude

chamber at the plume location. The placement of various filters into the field of view permits wavelength calibrations and resolution checks to be made, see Reference 2.

2.1.2 LWIR Rapid-Scan Spectrometers

The purpose of these instruments is to measure plume spectral radiances in the LWIR throughout the 6-24 μm wavelength range, and as such constitute the principal optical diagnostic system for this program. The desired resolution is about 0.5 μm with NEFD values of $\sim 10^{-10}$ watts/cm^2. To achieve such performance requires background cooling. In terms of the plume measurements in the Calspan test chamber, this requires not only cold ($\sim$80°K) optics but the elimination of radiation from the opposite wall of the chamber. This can be accomplished by either wall cooling (cryopanels) or by the use of confocal optics, whereby the field of view (FOV) is returned back on itself from a cooled mirror. For this reason, and for purposes of economy and overall feasibility, it was decided to design the instruments in terms of two components, one of which covers the range from 6 to 12 μm, and the other from 12-24 μm. These instruments each use a CVF and operate basically in the manner previously described for the SWIR scanning spectrometer, see Figure 4a. Because of thermal stress problems associated with cooling a composite CVF (6-12-24 μm) to liquid N_2 temperatures, and further to spin the filter at the required 12,000 rpm, it was decided during the preliminary design stage to separate the wavelength coverage as mentioned above. Critical design criteria are considerably lessened, spin speeds are lower, the CVF substrates are each made of a single homogeneous disc of material, and finally, by deploying the instruments as shown in Figure 2 the cooled wall problem is eliminated. Each optical system looks into the other, at no time viewing a surface that has not been cooled to liquid nitrogen (LN_2) temperatures.

An optional modification of the instruments is employed in the test program, wherein the CVF discs are replaced by broadband, stationary cold filters. These filters can be specified to match specific wavelength band-passes; i.e., the specific $\Delta\lambda$ of interest to the FAIR flight program. The

rocket thrusters involved in that flight program have large expansion ratios, and the resultant plume exit plane temperatures are extremely low. Possible freezing of vibrational modes of the product molecules is expected to dominate the emission spectra from the near field of these motors. These intensity levels are anticipated to be too low to permit spectral scans to be obtained. Thus, the use of fixed filters may be employed to assess the plume radiance in specified wavelength intervals.

The principal design problem involved the coupling of the instruments to the test chamber in a manner to eliminate or minimize possible cryopumping of rocket exhaust gases onto the cold optical surfaces inside the spectrometer. The use of additional windows was considered and include such associated problems as cost, possible breakage and the need to cyclically heat and cool the windows to remove any deposit buildup. Alternatively, the use of a fast-acting valve to close off the spectrometers from the tank environment within some 50 msec of the data scan time interval proved to be more tractable.

Figure 4b shows various details of the optical system in its present configuration, with the CVF discs replaced by broadband filters. The spectrometer (used as a radiometer for application to the FAIR thrusters) is coupled to the vacuum chamber by a 4 inch diamet r, remotely operated gate valve G. This forms the primary seal between the high vacuum in the spectrometer chamber and that of the test chamber. In addition, a fast-acting shutter-valve, FS, is also used to minimize possible degradation of the cold optics due to cryodeposition of test chamber gases. Mirrors M_1 and M_2 effectively image the detector D onto the centerline of the rocket nozzle with a magnification near 4.5. The detector area is 1 x 1 mm, and the system thereby provides a spatial resolution of about 5 x 5 mm at the thruster centerline. The mirrors and all elements in the optical train are cooled to liquid nitrogen temperatures. These elements include a shutter S, an Irtran window W, 2 x 2 mm baffle B, and the filter F. The detector is Ge:Hg* which, together with the associated MOSFET and load resistors, is mounted at the end of

*Aerojet Electrosystems, Azusa, California

a liquid He cooled Heli-Tran* unit, HTL, which provides high-capacity heat sink cooling. The unit has a thermocouple temperature readout, feedthrough for detector leads, and a separate vacuum line. The vacuum shroud, VS, is thermally coupled to the LN_2 source by a strap. It was found in operation, however, that the entire assembly between the vacuum shroud and the mirror M_2 had to be completely enclosed and thermally very closely coupled to the LN_2-filled inner liner. This was needed to prevent radiation from the instrument housing (at room temperature) from reaching the detector by reflections. Considerable effort was expended to achieve the low background signals required for the rocket tests.

As discussed previously, two units such as described above are deployed across a major diameter of the test chamber, see Figure 2. The second utilizes a Si:As detector to span wavelengths to 24 μm. Primary alignment has been completed, in which the centers of the M_1 mirrors and a reference point on the rocket nozzle centerline have been made colinear. This reference point is established to within ± 0.2 mm, and is physically reproducible by means of a special fixture. All optical focusing and FOV alignment, as well as the placement of the rocket nozzle, is made relative to this reference point.

Figure 5a and 5b shows photographs of one of the LWIR spectrometers installed on the 10-foot-diameter test chamber. Each unit has outer dimensions of approximately 30 inches in length and 10 inches in diameter. The view shown in Figure 5a depicts the quick-operating gate valve, as well as the liquid helium load and vent lines for the Ge:Hg detector mounted on the Heli-Tran unit seen at the top of the assembly. Figure 5b portrays a view of the opposite side of the same spectrometer assembly, showing the large vacuum line from the central pump unit which is located at one end of the test chamber. This central pump station is used to evacuate both spectrometer assemblies through dual vacuum lines such as shown in Figure 5b.

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*Air Products and Chemicals, Inc., Allentown, Pennsylvania

Preliminary tests have been conducted to assess the problem of cryodeposition on the cold optics. The end flange of the spectrometer and the mirror M were replaced by Plexiglass, the system was evacuated, cooled and exposed to the test chamber by opening the main gate valve G. Even with the tank pressure deliberately raised with moist air to 10 μ Hg, there was no visible evidence of any frosting over a 15-minute interval. On the basis of these preliminary tests, it was felt that cryodeposition for small thrusters such as will be employed during the FAIR tests will not be a problem, and would not even require the use of the fast-acting shutter valves.

An intensity calibrating system was installed in the test chamber. The selection of a suitable technique was based on past experience with the similar SWIR scanning spectrometer discussed previously. It is essential to be able to calibrate the spectrometric systems <u>in situ</u>, under vacuum conditions on a daily basis. The tank environment, however, especially when solid propellants are tested, is extremely hostile to this application. Any calibrating radiation source must be separately and completely shielded and evacuable to present contamination and an attendant emissivity change. For the present system, provision was made to sweep a series of wires of known diameter through the optical beam. By using wires with diameters as low as 10^{-3} inch, at a location where the beam is 3 in. diameter, a very small effective source area is obtained. This is needed to provide low radiance levels from the room temperature ($\sim$300 °K) wire surface. The principal advantage of this technique is that by having the wires at the ambient tank temperature, the radiation is representative of a 300 °K blackbody, independent of wire emissivity. The system is thereby independent of contamination effects, and final precision rests simply on a measurement of tank wall temperature and the assumption of thermal equilibrium of the wires and the inner tank walls during the test.

Figure 6 shows early results from this radiometer radiance calibration system. The signals resulting from the passage of the spokes and the thin, calibrating wires are shown. After the system was modified to reduce the effective background temperature, the final signal to noise ratio was about 3X better than shown by these records.

The application of the LWIR spectrometer system and calibration procedure to a specific rocket motor configuration is presented in Section 2.4, in which the hydrazine thruster experiments are discussed.

2.2 PARTICULATE FLUX DIAGNOSTIC

The particulate collection and analysis techniques employed for selected tests are based upon the experimental techniques developed during the HIT motor test series and described in detail in Ref. 1. Four individual assemblies are used to collect particles emitted from the rocket motors under investigation. These collectors may be located at different angles with respect to the rocket motor axis; for example, to ensure that statistically measurable samples are acquired for each fuel-nozzle configuration. Previous experience(1) has shown that the measured particle densities (measured at a given point in the flow field) can be used to determine the total particle flux from the motor. Thus, for a given test, the collector assemblies could be located as follows: One collector assembly on the motor axis and three others at angles of about 30°, 50°, and 70° off the motor axis. A two to three order of magnitude variation in particulate flux can be simultaneously collected and analyzed without approaching saturation from particle laden propellants or poor statistical samples from relatively particle free propellants.

One of the particle collector assemblies is shown schematically in Figure 7. The particles enter through the large aperture of the Graflex 1000 shutter shown at the left of the figure. There are three glass collector slides located within the assembly on which to collect the particles that have traveled into its interior. The radially located slide is to be compared to the two control slides, located so as to be parallel to the undisturbed-flow stream. Those particles that enter the collector assembly must pass through a shock wave which is formed by exhaust gases in front of its entrance aperture. In addition, their momentum must be sufficient to enable preferential collection along radial lines from the rocket exit.

Each collector assembly can be separately operated and monitored to enable sampling procedures in a wide variety of sequences. One typical procedure is to sample the exhaust flow stream immediately after the thruster chamber pressure has reached a maximum. The shutter at the entrance to the collector, see Fig. 7, will open at this point in time and close just prior to the time when the flow stream is interrupted by reflected gases from the end walls of the high altitude test chamber. The collected sample can then be analyzed to determine particulate size and number density as a function of radial and axial position.

Figure 8 shows a typical oscilloscope record of a calibration test. To obtain this recording, a photomultiplier was placed within the particle collector and directed so as to record the light intensity from a broad source located just outside the Graflex 1000 shutter. As the shutter opens, the recorded photon flux is directly related to the open area of the shutter. The upper trace in the figure shows this output as a function of time after a trigger pulse has been applied to the shutter solenoid. The shutter starts to open 4.0 ms after trigger application, and it is fully closed about 2.2 ms later. This signal increase, maximum and subsequent decrease is linearly proportional to the photon flux entering the collector and is also a measure of the time over which particulate matter will be collected during the thruster tests. The second trace on the oscilloscope record presents a signal that is related to a switch contact closure that occurs after the mechanical shutter is in motion. This is actually the set of contacts used to ignite "M" type flash bulbs when using the mechanism in its normal configuration. The time between contact closure and shutter opening is constant and subsequently used to verify the time during which particles are collected. Timing calibrations have been obtained for each of the four collector assemblies, for shutter speeds from 1/60 to 1/500 second.

The particles are collected on glass slides covered with a thin film (<< 1000 Å) of Collodion, see Fig. 7. The film is obtained by casting a mixture of 1% Collodion in amyl acetate onto a water surface and picking it up on the slide. After an air dry, four other materials are attached using

additional Collodion as an adhesive. These materials present alternate surfaces on which to collect particles having different characteristics. The four materials placed on each collector slide are (1) a 1/8 in. diameter by 0.050 in. thick soft copper disc with a chemically etched surface, (2) a 5/16 in. diameter by 0.005 in. thick soft copper disc with a chemically etched surface, (3) a strip of 0.001 in. thick copper conducting tape (1/8 in. x 1/4 in.) with a polished surface, and (4) a 1/8 in. diameter carbon coated Formvar grid manufactured for electron microscopes by the Ernst F. Fullam Company.

Each of the collected samples is studied using the Calspan materials laboratory equipment. The main tools used include a Bausch and Lomb Research Metalograph (resolution $0.5 \mu m$), a Phillips 100 B electron microscope (magnification of 10^3 to 6×10^5) and an ETEC-Autoscan scanning electron microscope. The combination of this equipment with the several alternate collecting surfaces enables significant samples and proper analysis to be obtained for specifying the particulate flow field.

Figure 9 shows a photograph of the four individual collector assemblies installed inside the high-altitude test chamber. The collectors are shown arranged on a mounting rig, one in each quadrant equidistant from the rocket thrust centerline, prior to a test of one of the small FAIR rocket thrusters. Data obtained from these assemblies during the hydrazine thruster tests is presented in Section 2.4.

2.3 ROCKET MOTOR DESIGN AND PROPELLANT COMBINATIONS

This subtask concerns the rocket technology necessary for successful completion of the planned experimental test program. This includes: the various rocket systems such as propellant feed lines, ignition system, motor diagnostics, etc.; computations as to the optimum propellant combinations, mixture ratios, gaseous simulation techniques, solid propellant burning areas, etc; and the finalization of the type of nozzle configurations that will be utilized as part of the experimental screening program.

In general, rocket engines of Calspan design will be employed for any liquid bipropellant test firings as part of the overall screening tests. To minimize the complexities involed in the handling of corrosive and toxic propellants, two techniques are being developed. One utilizes only small, premeasured amounts of propellants in capsule form, suitable for the short-duration test technique employed at Calspan. The other technique utilizes reacting gaseous propellants which can be substituted for the liquids to produce equivalent combustion chamber conditions. The rockets employ heat sink cooling (because of the short firing duration, < 100 ms), and are constructed so as to provide for ready interchangeability of nozzles and injectors between units. It is presently planned that bipropellant rockets will utilize both conical and contoured nozzle configurations for various exit area ratios.

The monopropellant hydrazine thruster assembly (including catalyst bed) used for the FAIR II tests, was obtained from Philco-Ford/Aeronutronic and modified as required for use in the Calspan test chamber. This particular rocket configuration is low thrust (≈ 5 pounds) with a large area ratio conical nozzle.

Solid propellant rockets to be employed in this program are of Calspan design: A typical motor, developed for the ARPA program discussed earlier[2], is shown in Figure 10. The unit consists of a cylindrical combustion chamber, propellant web inserts, bolt-on nozzle, nozzle closure diaphragm, and ignition system. The propellant is loaded into the combustion chamber on reusable metal propellant holder inserts; virtually any grain geometry (e.g., wedges, star, cruciform, concentric cylinders, all of which have previously been employed by Calspan) can be provided by this technique. Since the required burn time is quite short (nominally 100 ms or less), the propellant web thickness need normally be only 0.030 - 0.050 in. even for comparatively high burning rate propellants. As a result, the motor propellant load is not cast or extruded in the conventional manner, but rather is simply made up by cementing thin strips of propellant directly to the metal propellant holder. These grains, several of which can be fabricated at a single time, allow for rapid combustor loading, simplified propellant handling, and low motor costs.

The nominal solid-propellant surface (burning) area is calculated to provide the desired steady-state chamber pressure, based on ballistic properties supplied by the manufacturer. Present plans call for a chamber pressure in the range of 500 psia, with a thrust level on the order of 500-1000 lb. Final adjustments in the propellant burn area to match the desired operating conditions are readily made based on the results of actual checkout firings.

Bolt-on, uncooled nozzles are attached to the combustion chamber; although only a conventional convergent-divergent nozzle happens to be shown in Fig. 10 it is planned that a throttling pintle nozzle configuration of the type illustrated in Fig. 11 will also be employed in this program. This nozzle design will be such as to allow the pintle to be located at a number of axial locations so as to simulate throttled, reduced thrust operation. Based on field of view requirements for the optical measurements, and general ease of fabrication and handling, a nozzle exit diameter on the order of 5-6 in. will be used for both the conventional and pintle-type nozzles.

To provide the rapid combustion pressure rise time required for short-duration rocket firings, a gaseous ignition technique is employed rather than conventional squibs or igniters. Briefly, the technique is as follows:

After the propellant is installed, the combustion chamber is sealed off at the nozzle end with a plastic closure sized to rupture at the design chamber pressure. Just prior to firing, the combustion chamber volume is filled to a low pressure (on the order of a few psia) with an oxygen-rich H_2/O_2 mixture ($O/F \approx 20$). Ignition of this mixture with a spark plug instantaneously exposes the entire propellant surface to a high temperature ($\sim 5000°F$) oxidizing environment, resulting in uniform ignition of the entire propellant surface in a very short time. Following ignition, the combustion products are contained in the combustion chamber until the nozzle diaphragm ruptures at the design pressure. With a correctly sized propellant burning area, the combustion chamber pressure remains constant and a steady flow exhausts through the nozzle until the propellant is consumed.[2]

Solid propellants anticipated for use in booster motors will be investigated, including CTPB and HTPB type propellants, with and without aluminum additive to evaluate the effects of aluminum on the IR plume signature. Equilibrium calculations for typical propellant forumations are given in Table I.*

2.3.1 Gaseous Equivalent Technique

The simulation of reactive liquid bipropellants by means of combusting gaseous equivalents has been successfull employed by Calspan for many years in short-duration rocket base heating and propulsion system performance studies. The use of gaseous reactants has been found to greatly simplify the propellant supply system; minimize propellant quantities required for individual firings; avoid the usual problems and complexities involved in handling and transfer, pressurization, metering and injection, etc. of liquid propellants; and ensure uniform and complete chemcial reaction in the rocket combustion chamber. It should be emphasized that the exhaust plume formed from such a gaseous rocket is referred to as an ideal plume; that is, the components are uniformly mixed, there are no propellant droplets on the walls, etc. It is planned to employ this gaseous equivalent technique to simulate the combustion of N_2O_4/N_2H_4, for example, and compare plume radiance measurements with those obtained during actual liquid tests. A brief discussion of the thermodynamic and chemical considerations involved in this technique is presented in the following paragraphs.

Based on the rigorous thermodynamic reasoning detailed in Ref. 6, only two conditions have to be satisfied to completely duplicate the combustion reaction of any liquid bipropellant combination by the combustion of an appropriate gaseous fuel and oxidizer which are initially at the same initial temperature. These are:

(1) The elemental composition (i.e., atomic species and their relative proportions) of the liquid and gaseous combinations must be the same;

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* Computations using NASA/LERC Chemical Equilibrium Code were provided by Dr. R. Rhoades of USA-MICOM.

(2) The initial energies (i.e., total sensible and chemical enthalpies) of the two propellant combinations must be identical.

Although condition (1) is normally easily satisfied, condition (2) involves a trial-and-error approach to arrive at a suitable mixture of gaseous components.

To briefly illustrate the procedure employed, consider the common earth storable combination N_2O_4 (nitrogen tetroxide) oxidizer and MMH (monomethyl hydrazine, $N_2H_3[CH_3]$), reacting at a mixture ratio of 1.6 (oxidizer/fuel, by weight) at a chamber pressure of 625 psia. A simple chemical balance shows the atom ratio of the reactants to be N:O:C:H = 1:1.02: 0.238:0.119, by weight, at the specified mixture ratio of 1.6. The readily available, commonly used N_2, O_2, and H_2 gases are convenient initial choices for providing the N, O, and H atoms in the gaseous reactant mixture. Many types of gases will provide the required carbon atoms including, for example, any hydrocarbon gas, carbon monoxide or carbon dioxide, cyanogen (C_2N_2), etc., all of which contribute C and another element of the type already present. By a trial and error process (via an estimate of the total chemical energy available), the various gases can be eliminated. In the present example, ethylene, C_2H_4, was found to be the best choice. Upon satisfying the atom ratios, by means of a 52% N_2 - 48% O_2 gaseous oxidizer and 80% H_2 - 20% C_2H_4 gaseous fuel (both percentages by volume), equilibrium combustion chamber calculations can be made, giving the results shown in Table II. The close duplication of the gas species and important thermodynamic parameters is obvious. Such a mixture was successfully used by Calspan, both in a recent sub-scale program performed for NASA/MSFC involving simulation of the Skylab attitude control rockets (Ref. 7) and in a plume-mapping investigation for the Boeing Company involving the Minuteman III post-boost propulsion system axial engine (Refs. 8 and 9).

A similar rationale as that just described will apply to the generation of a gaseous equivalent for the N_2O_4/N_2H_4 liquid system. Basic mixtures of N_2, O_2, and H_2 can be considered, modified as required with other gases such as, for example, N_2O, NH_3, NO, etc. to produce the proper combustion energy release. Similarly, interhalogen-hydrazine combinations may be simulated by appropriate F_2, O_2, N_2 and H_2 basic gaseous mixtures, modified as required by the other gases such as NH_3 and NF_3.

Detailed thermochemical calculations for a number of liquid propellants of potential interest for system applications were provided to Calspan by Dr. R. Rhoades of MICOM and expanded using the NASA/LERC Code, which subsequently became operational on the Calspan 370/165 computer. The computations included data for CTF and CPF/hydrazine and N_2O_4/hydrazine propellants.

Based on the thermochemical calculations for CTF/N_2H_4 propellants as a function of mixture ratio, it was decided that the Calspan tests should be performed at O/F ratios corresponding as closely as possible to maximum specific impulse (I_{SP}) and peak density-impulse (ρI_{SP}), since these optimums would most likely be prime design points for future operational applications. These parameters are plotted in Fig. 12 for both CTF/N_2H_4 and N_2O_4/MMH propellant combinations. It can be seen that for the CTF/N_2H_4 combination, simultaneous maximums occur for both I_{SP} and ρI_{SP} at an O/F $\approx$ 3. Similarly, an O/F of 1.5 provides dual maximums for the N_2O_4/MMH propellant combination; these mixtures ratios will be employed during the test program.

Thermochemical calculations were then made to establish appropriate gaseous equivalents for these liquid propellant combinations. As seen in Tables III and IV, an oxidizer consisting of mixture of gaseous Cl_2 and F_2, when reacted with a gaseous fuel consisting of NH_3 and H_2, provides an excellent duplication of the CTF/N_2H_4 system. Similarly, gaseous N_2/O_2 burning with H_2 provides a good duplication of the N_2O_4/N_2H_4 propellant system.

Additional calculations have also been made in an attempt to improve the N_2O_4/N_2H_4 system gaseous equivalent by adding N_2O and NO to the N_2/O_2 oxidizer. In all cases, the combustion temperature was excessively low, and the originally selected $N_2/O_2 + H_2$ combination still provides the closest simulation (within $\approx$ 4% on combustion temperature). As a result, this combination will be employed for the N_2O_4/N_2H_4 simulation tests.

2.4 FAIR-PIE THRUSTER EXPERIMENTS

A technology interchange meeting was held at ABMDA, Washington, concerning the Plume Interaction Experiments (PIE) to be flown on FAIR II with a discussion of possible laboratory measurements to be performed at Calspan in support of the program. The laboratory measurements would be directed at determining the plume near-field IR radiation from the PIE-type thrusters. There are three small thrusters under consideration: a 5-lb thrust monopropellant hydrazine motor, a 5-lb thrust peroxide decomposition thruster, and a 40-lb thrust solid motor utilizing a particulate-free solid propellant. Small thrusters of each type will be tested in the high-altitude chamber and measurements of near-field plume IR radiation obtained.

2.4.1 Monopropellant Hydrazine Thruster Tests

The purpose of the experimental tests was to obtain measurements of the near-field core radiation in the 10μm region of the spectrum. This radiation arises from transitions in NH_3 which constitute approximately 40 percent of the rocket exhaust species. The measurements can then be utilized to verify the predictions provided by AERL using their plume radiance model. Diagnostics were also provided to assess the possibility of particulate matter in the plume emanating from the catalyst bed.

A hydrazine thruster of the type to be used on the FAIR II-PIE flight experiment was received from Philco-Ford (P-F), the PIE prime contractor, for use in the FAIR II support experiments at Calspan. The thruster, originally developed by Hamilton Standard, produces a thrust of $\approx$ 5 pounds at a chamber pressure $\approx$ 100 psia. As delivered to Calspan, the thruster assembly consisted of a thrust chamber containing Shell 405 catalyst, a conical nozzle, a fast acting solenoid valve, integral line filter, and necessary connecting tubing and mounting provisions. An outline sketch of the motor showing pertinent dimensions is presented in Figure 13.

During the actual PIE flight experiments, a long sequence of pulses (each of approximately 150 ms duration) are planned; the first 20-30 pulses are employed to bring the thruster and catalyst bed to optimum operating conditions. The normal rate of rise for a cold hydrazine thruster is far too long for the available test time in the altitude chamber (on the order of 150 ms relative to a test period of only ≈ 15 ms). After receipt of the thruster from P-F, Hamilton-Standard (H-S) technical personnel were contacted to explore techniques by which the thruster might be modified to include a possible divert capability necessary for providing simulated hot catalyst bed, "n-th" pulse firings in the Calspan vacuum facilities. Out of these discussions emerged the approach employed for the test series; namely, artificial electrical heating of the catalyst bed to temperatures equivalent to those obtained during a normal engine firing sequence with flowing propellant. By this approach, the catalyst bed can be initially thermally conditioned to the temperature corresponding to any desired pulse and then fired (i.e., propellant flow initiated) directly into the chamber without the necessity of first flowing hydrazine propellant through the bed to warm it to operating conditions. Heating of the bed by means of actual hydrazine flow would require an overboard venting of the resultant exhaust products outside the altitude chamber to avoid undesirable increases in the ambient pressure level before initiation of data-collecting firings into the chamber.

To determine the appropriate temperature to which the catalyst bed should be preheated to simulate any particular pulse of interest, a static firing sequence would first be required during which bed temperatures would be determined. Based upon this information, a heater capable of heating the bed to the same temperature would then have to be developed, followed by further test firings to verify proper simulation. Because of Hamilton-Standard's familiarity with the motor and readily available hydrazine rocket test facilities, it was decided that this developmental effort could be handled more expeditiously by H-S than by Calspan. As a result, a subcontract was awarded to H-S to perform two tasks in support of Calspan's FAIR II thruster tests. Briefly, the H-S effort consisted of (1) development of a catalyst bed heater which would allow proper warm bed, "n-th" pulse simulated firings, and (2) fabrication and checkout of a hydrazine propellant supply system for use with the thruster at Calspan during the FAIR-II tests. The complete Hamilton-Standard Work Statement is presented in Appendix A of the present report.

2.4.1.1 Thruster Operation

Rocket Assembly and Propellant System -- The hydrazine thruster is depicted schematically in Figure 13. A stainless steel propellant line leads from the motor fuel valve (not shown) to a series of fine capillary tubes which distribute the fuel (N_2H_4) uniformly through the catalyst bed. The bed consists of a fine mesh of Shell 405 catalyst. Upon contact with the catalyst, hydrazine decomposes into its primary exhaust products of H_2, N_2, and NH_3. This rocket is rated as having a nominal 5-lb thrust at a chamber pressure of 100 psia.

Three thermocouples were located as shown in the figure: one on the nozzle wall, one on the motor housing near the catalyst bed, and one on the propellant manifold (engine mounting flange). These thermocouples were used to monitor temperatures throughout the thruster firing sequence. Each pulse of the motor ($\sim$ 150 ms duration) raises the catalyst bed temperature because of the energy being released by the decomposition of the propellant. A thermal standoff barrier is provided between the catalyst bed and propellant manifold to preclude the possibility of preheating and vaporizing the propellant before it enters the catalyst bed.

Figure 14 is a schematic of the thruster and propellant supply system as installed on the high altitude test chamber. The propellant system was mounted on the outside bulkhead of the test chamber with a feedthrough line to the motor solenoid valve. A typical engine firing sequence was as follows. The test chamber was evacuated to a pressure of $\sim 0.5\,\mu$m; this pressure level was dictated by the optical and particle collector diagnostic procedures. After the desired pressure was obtained, the appropriate fuel system valves were opened and the propellant lines were pressurized to 250 psia. The main propellant valve was triggered electronically from a remote fire switch and remained open for the predetermined pulse time. Both optical and particle collection functions were sequenced automatically from this fire switch. It was observed that a single 150 ms pulse increased the pressure in the test chamber to approximately 20 μm.

Performance Characteristics -- Figure 15a shows data records for a characteristic sequence of events for a single firing. The lower trace is the applied voltage to the motor valve and indicates the pulse duration of approximately 150 ms. The upper trace is a record of the chamber pressure, which attains a value near 80 psia at the end of the pulse. The shape of the pulse and the delay from applied valve voltage to P_c rise varied from pulse to pulse as a function of catalyst bed temperature. The record shown in Figure 15a was a room temperature, "cold" pulse with a delay time on the order of 45 ms. The pulse shape and relatively slow pressure rise was characteristic of a cold pulse. The additional "spike" on the trace shows the time of particle collector shutter opening. A preset delay from the applied valve voltage (i.e., firing switch) determined this time.

Figure 15b shows data records for a hot pulse with a catalyst bed temperature of 900°F. The delay was on the order of 20 ms with a rapid P_c rise to a uniform value of 110 psia.

In the flight configuration, approximately 30 pulses of a similar thruster were made to clear the catalyst bed of fine particles that might be exhausted in the plume, and to bring the motor to a more effective operating temperature. An experimental 30-pulse sequence was performed at both Hamilton Standard and Calspan. Thermocouple temperature-time histories obtained during the sequences are presented in Figure 16. Shown are the catalyst bed and nozzle wall temperatures (see Figure 13 for thermocouple locations), as a function of motor firing pulse number. The noticeable difference between the two bed temperature profiles was attributed to the different techniques that were used to attach the thermocouples to the motor assembly. At Hamilton Standard, the thermocouple wires were inserted in a metal sleeve soldered closed at the end. This was then fastened to the motor with metal straps. At Calspan, the thermocouple leads themselves were spot welded directly to the motor at the designated locations. The lower temperature profile indicated by the Hamilton Standard installation may be at least partially attributed to a thermal lag caused by the metal sleeve. During this 30-pulse sequence, it can be seen that the bed temperature increases to approximately 1100°F because of the energy

release during the decomposition reaction. In both pulse sequences, the nozzle wall temperature change was small, with the nozzle attaining a final temperature of 150°F. The motor mounting flange remained at room temperature throughout the sequence.

As discussed earlier, a single pulse of the thruster increased the vacuum facility pressure to about 20 micrometers. Heating of the catalyst bed by actual hydrazine flow (similar to the 30-pulse sequence shown in Figure 16) would require an overboard venting of the exhaust products to avoid excessive increases in the ambient pressure level above that required for proper optical and particulate diagnostic operation. The approach employed at Calspan utilized artificial electrical heating of the catalyst bed to temperatures equivalent to those obtained during a normal engine pulsing sequence. With this approach, the catalyst bed was initially thermally conditioned to the temperature corresponding to any desired pulse number in the 30-pulse sequence.

The longer time required for artificial heating of the catalyst bed, however, gave nozzle wall temperatures higher than those measured during a natural heating sequence, due to heat conduction processes to the nozzle wall. For example, when the bed was heated artificially to 600°F, the nozzle wall thermocouples indicated a temperature of approximately 300°F, while the corresponding nozzle wall temperature for a natural pulse sequence was 100°F, see Figure 16. A water cooling coil was affixed to the nozzle so that the nozzle wall temperature could be controlled. Optical plume measurements were made with both a cold and hot nozzle for a given catalyst bed temperature. No significant effect of nozzle wall temperature was observed for the radiance measurements. Numerous pulses of the rocket motor were made at various catalyst bed temperatures and motor positions with respect to a fixed optical reference point. Optical data were obtained over a catalyst bed temperature range from 72°F to approximately 900°F, and nozzle wall temperatures were controlled from 72°F to 350°F.

Figure 17 shows the thruster assembly mounted in the test chamber. The chamber pressure transducer is seen mounted on top of the motor and the water coolant jacket is shown over the exhaust nozzle. The heater can be seen clamped to the catalyst bed section of the assembly.

Summary -- The rocket and propellant supply system gave reliable operation and consistent day-to-day reproducibility of pulse conditions and overall performance. Initial difficulty was experienced with the bed heaters shorting out while simulation of the higher operating temperatures was being attempted. This was resolved by adding an isolation transformer to the heater circuit to prevent this occurrence. The electronic timing sequence controlling the optical and particle collector was established and employed effectively.

2.4.1.2 Plume IR Radiance Measurements

The basic question concerning the hydrazine thruster was whether the near-field radiance from NH_3 in the plume will affect, or possibly degrade, the desired far-field measurements of interest in the FAIR-PIE flight experiment. Preliminary calculations of this near-field radiance have been made, but have been shown to depend critically on the relaxation kinetics of NH_3 during the expansion processes in both the rocket nozzle and the near-field plume. The effective vibrational temperature of the ammonia molecule has been calculated to lie in the range from 200°K to much less than 100°K, depending on the degree to which the pertinent vibrational mode is equilibrated during the overall expansion process. Thus, a primary objective of the experiments at Calspan was to obtain radiance measurements to aid in substantiating the kinetic models used in predicting plume near-field radiance.

Extensive calculations had been made at AERL on the anticipated range of plume radiance values for the hydrazine thruster, as a function of NH_3 rotational and vibrational temperatures. These predictions have been used to provide baseline radiance levels to which the optical system has been configured. The bulk of the radiation appears in the wavelength interval between 9 and 12 μm. Because of the generally weak radiance levels associated with the low nozzle exit

plume temperature, it was found infeasible to attempt to spectrally resolve the radiation. Thus, as discussed in Section 2.1, the scanning spectrometers were modified for use with fixed filters. The primary measurement used a bandpass between 7.97 and 12.23 μm. With this sytem, calculations have indicated that measurements could be made as far as four nozzle radii downstream. The experimentally determined axial distribution of plume radiance can then be compared with the predicted values to enable an assessment of NH_3 vibrational temperature and verification of the kinetic model.

Except for the problem of obtaining a sufficiently low background for the radiometer, the optical measurements on the hydrazine motor proceeded as planned. The recording oscilloscope was triggered from the fuel-valve switch-on, (as discussed in the preceding section), and traces obtained at various sweep rates and gain settings. A typical trace is shown in Figure 18, in which the IR radiance level can be directly compared with the corresponding chamber pressure history. A steady-state test time of 30-40 ms can clearly be distinguished. The subsequent radiance rise results from plume gases in the field of view which have reflected from the end of the test chamber. Although for some conditions, an initial overshoot was observed at motor turn-on, the test time interval was always sufficient to obtain a steady-state radiance level.

Tests were conducted at three stations downstream of the rocket motor, at 0.30, 1.05, and 1.80 in. measured from the motor exit plane. The results of these tests are shown in Figure 19. The figure shows both calculated (AERL)[10] and experimental (Calspan) values of the total in-hand NH_3 radiance in units of watt cm^{-2} sr^{-1} as a function of downstream distance in inches. The results indicate that an adequate radiance prediction must include at least a finite-rate model of vibrational relaxation. For example, a prediction employing a vibrationally frozen NH_3 model yields a constant radiance level with downstream distance.

The data shown in Figure 19 were calibrated by comparing the radiance from the gas with that of 7.0×10^{-4} in. diameter wire, placed in the instrument FOV. A non-linearity of signal as a function of wire diameter was found, and is presently under investigation. Both electronic and non-ideal optical configurations are being examined as a likely source of the non-linearity of calibration data. Scaling with a number of equal-diameter wires is exact, however, and this feature has been incorporated in a modification to the calibration system. The net effect of the present non-linearity is an uncertainty band in the radiance of 50 percent measured upward from the data plotted in Figure 18. This possible shift would result in a more favorable comparison with the AERL finite-rate, boundary-layer corrected prediction. This limitation affects only the absolute radiance levels; the relative values as a function of downstream distance are correct, and result in a favorable comparison with the finite-rate, boundary layer corrected radiance model.

2.4.1.3 Particulate Flux Measurements

The particle collectors discussed in Section 2.2 were used to investigate the possible presence of particulate matter in the exhaust plume from the small hydrazine thruster. The collectors were placed at predetermined distances downstream from the thruster nozzle exit plane, and the shutter opening period was adjusted to occur at predetermined times. Thus, the possible occurrence of particulate matter as a function of both radial and axial distances from the thruster could be correlated with specific time intervals during decomposition and chamber pressure buildup. A typical flow assembly configuration can be seen in Figure 9.

Two series of tests were performed in which the particle collectors were the primary diagnostics: (1) measurements prior to subjecting the hydrazine thruster to a standard vibration schedule as described in Appendix A, and (2) measurements on the first series of pulses immediately following the vibration sequence. The results are presented in Tables V and VI, respectively. The pre-vibration tests (Table V) were performed at various catalyst bed temperatures which are listed in column 3 of the table. Column 4 shows

pictorially the shutter-open period as correlated with real-time measurements of the thruster chamber pressure. The particle collector locations and observed results are given in the last column of the table. For example, collector number 2 sampled the exhaust flux from the hydrazine thruster with the catalyst bed at room temperature prior to opening the propellant valve. This number 2 collector was located axially 62 inches from the nozzle exit plume and 14 1/2 inches radially outward from the thrust centerline. A few solid particles were evident, and the many signatures indicate that vaporizable particles had impacted the collector slides. It should be noted that, for this particular test, the shutter-open time was sequenced to occur early in the pulse before substantial decomposition.

A preliminary analysis of all experiments indicates that very few solid particles were evident in the exhaust plume. However, signatures from vaporizable material such as solid or liquid hydrazine were evident when the catalyst bed temperature was near 300°K. The particles produced holes in a thin carbon-coated Formvar surface, but had apparently vaporized by the time the test chamber was recycled and the particle collectors removed for analysis.

Section 3
CONCLUSIONS

This final report summarizes progress on the laboratory measurements phase of the Plume Interference Assessment and Mitigation Program. There are two main tasks: the first involves a series of rocket propellant screening experiments in which near-field IR plume radiance characteristics are investigated, and the second task is to obtain laboratory measurements of plume radiation from small thrusters of the type to be flown on the FAIR II plume interaction flight experiments.

During the reporting period, effort has been expended primarily on the several subtasks required prior to extensive rocket tests, with initial measurements obtained on the first FAIR thruster. The main subtask involved the development of the LWIR diagnostic system. The design, fabrication, calibration and installation of two cooled LWIR scanning spectrometers have been completed for the high-altitude test chamber where the rocket motor tests are conducted. These instruments each employ a rapidly spinning circular variable-filter (CVF); one instrument covers the wavelength range from 6-12 μm with a Ge:Hg detector, and the other unit covers the 12-24 μm region with a Si:As detector. All the optical components are cooled to liquid nitrogen temperature, and the instruments are mounted across the 10-ft diameter of the test chamber. Thus, each optical system looks into the other, at no time viewing a surface that has not been cooled to LN_2 temperature.

A novel calibration technique has been developed for use with this diagnostic system. It is essential to calibrate the spectrometer systems <u>in situ</u>, under vacuum conditions on a daily basis. For the present system, a calibration wheel with thin wires as spokes sweeps a series of wires of known diameter through the optical beam. The principal advantage of this technique is that by having the wires at the ambient tank temperature, the radiation is representative of 300°K blackbody, independent of wire emissivity.

A second subtask involved the fabrication and calibration of additional particle collectors, utilizing the experience gained with a single collector on a previous plume diagnostic program. Four individually timed collector assemblies are now employed to obtain spatial data on plume particulates during a single rocket test.

The third subtask involved the rocket technology necessary for successful completion of the experimental test program. This included rocket motor design and fabrication, propellant fuel systems, motor sequencing and data acquisition systems, and calculations of propellant type and mixture ratio. The screening program encompasses not only several types of solid propellants, but both monopropellant (N_2H_4) and bipropellant (N_2O_4/N_2H_4, ClF_3/N_2H_4) liquid and gaseous systems, as well as a number of nozzle configurations. The test series for the small FAIR type thrusters was formulated and flight-type motors were obtained and modified for laboratory tests.

The first test series of a small thruster of the FAIR-PIE type has been completed. Plume radiance measurements in the 8-12.5μm wavelength region were obtained for a small (5-lb thrust), high exit-area-ratio motor, using monopropellant hydrazine. The radiance data indicate that an adequate radiance prediction for this type thruster must include a finite-rate model of NH_3 vibrational relaxation and possible effects due to expansion of the nozzle boundary layer into the outer regions of the plume.

The FAIR type thruster experiments and overall propellant/nozzle screening tests are being completed under a separately funded ABMDA research program.

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Table I
SOLID PROPELLANT PROPERTIES

P_c = 3000 psia

	CTPB W/16% Al		HTPB W/O Al		Pintle Solid W/19% Al		Solid Cluster W/ 16% Al	
	Chamber	Exit	Chamber	Exit	Chamber	Exit	Chamber	Exit
Temperature	3555° K	1703° K	3035° K	1121° K	3790° K	2065	3555° K	1700° K
Area Ratio	----	32	----	26	----	35	----	32
Al_2O_3(L)	7.5%	---	----	----	9.1%	---	7.5%	----
Al_2O_3(S)	----	8.1%	----	----	----	10.0%	----	8.1%
CO	22.2	21.2	11.5%	5.6%	18.9	18.7	22.3	21.4
CO_2	1.7	3.2	12.0	18.1	1.6	2.5	1.7	3.1
HCl	13.7	15.9	18.2	19.0	12.4	15.7	13.6	15.9
H_2	25.9	28.6	5.4	10.6	24.4	27.0	26.0	28.8
H_2O	15.9	14.4	41.0	36.6	17.4	17.1	15.7	14.2
N_2	8.1	8.3	9.7	9.8	7.9	8.2	8.1	8.3
	71% AP 16% Al 10% CTPB		87% AP ----- 10.5% HTPB		65% AP 19% Al 14.5% Binder 1.5% Other		71% AP 16% Al 10.5% HTPB	

Table II
COMPARISON OF PEACTION PRODUCTS OF LIQUID AND GASEOUS PROPELLANTS

Combustion Chamber Pressure = 625 psia

	Liquid $N_2O_4/MMH^{(1)}$	Gaseous N_2-O_2/H_2-$C_2H_4^{(2)}$
Combustion Temperature	3124°K	3208°K
Molecular Weight	20.4	20.5
Specific Heat Ratio,	1.23	1.23
<u>Specie Concentration (Mass %):</u>		
N_2	42.0	41.2
H_2O	28.9	29.3
CO	18.6	18.1
CO_2	7.5	7.9
H_2	1.7	1.5
OH	0.9	1.2
NO	0.23	0.35
O_2	0.06	0.12
O	0.04	0.07
H	0.01	0.08

(1) $N_2O_4/N_2H_3(CH_3) = 1.6$, by weight

(2) $\underbrace{52\%\ N_2 - 48\%\ O_2}_{\text{by volume}} / \underbrace{80\%\ H_2 - 20\%\ C_2H_4}_{\text{by volume}} = 5.7$, by weight

Table III

GASEOUS EQUIVALENT CALCULATION FOR CTF/N_2H_4

CHLORINE TRIFLUORIDE/HYDRAZINE

P_c = 1000 psia; r = 3.0

	COMBUSTION CHAMBER			EXIT ($\epsilon \approx 35$)		
	LIQUIDS	(GASES) $Cl_2/F_2 + NH_3/N_2$	(GASES) $Cl_2/F_2 + H_2/N_2$	LIQUIDS	(GASES) $Cl_2/F_2 + NH_3/N_2$	(GASES) $Cl_2/F_2 + H_2/N_2$
T	3941°K	4138°K	4262°K	1166°K	1315°K	1415°K
M	23.7	23.3	23.0	25.2	25.2	25.2
γ	1.198	1.192	1.188	1.338	1.306	1.283
C*	5952 FPS	6162 FPS	6298 FPS	5952 FPS	6162 FPS	6298 FPS
	SPECIES ~ MOLE FRACTION			SPECIES ~ MOLE FRACTION		
Cl	0.083	0.094	0.101	0.001	0.005	0.010
ClF	0.000	0.000	0.000	0.000	0.000	0.000
Cl_2	0.000	0.000	0.000	0.015	0.014	0.013
F	0.016	0.024	0.031	0.000	0.000	0.000
H	0.026	0.038	0.047	0.000	0.000	0.000
HCl	0.108	0.094	0.085	0.173	0.171	0.167
HF	0.560	0.543	0.530	0.614	0.614	0.613
H_2	0.022	0.025	0.026	0.000	0.000	0.000
N	0.000	0.000	0.000	0.000	0.000	0.000
N_2	0.185	0.182	0.180	0.197	0.197	0.197

Table IV

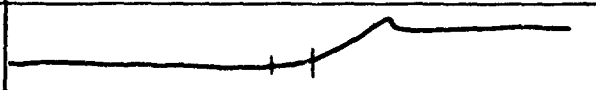
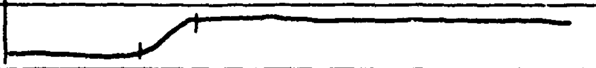
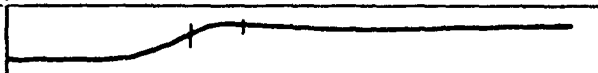
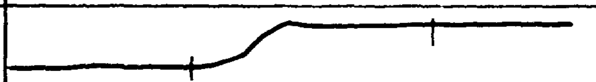
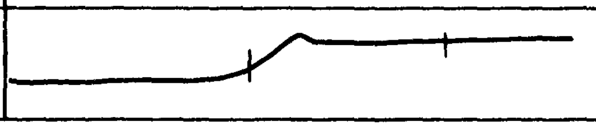
GASEOUS EQUIVALENT CALCULATION FOR N_2O_4/N_2H_4

NITROGEN TETROXIDE/HYDRAZINE

P_c = 1000 psia; r = 1.50

	COMBUSTION CHAMBER		EXIT ($\epsilon \approx 50$)	
	LIQUIDS	$N_2/O_2 + H_2$ (GAS)	LIQUIDS	$N_2/O_2 + H_2$ (GAS)
T	3284°K	3135°K	1154°K	1022°K
M	21.5	21.8	22.5	22.5
γ	1.148	1.153	1.271	1.282
C*	5788 FPS	5601 FPS	5788 FPS	5601 FPS
SPECIES ~ MOLE FRACTIONS				
H	0.009	0.008	0.000	0.000
H_2	0.046	0.032	0.000	0.000
H_2O	0.470	0.493	0.561	0.558
N_2	0.402	0.409	0.427	0.427
NO	0.011	0.012	0.000	0.000
O	0.004	0.003	0.000	0.000
OH	0.038	0.029	0.000	0.000
O_2	0.018	0.017	0.013	0.015

Table V
FAIR II HYDRAZINE THRUSTER
PRE-VIBRATION TEST DATA

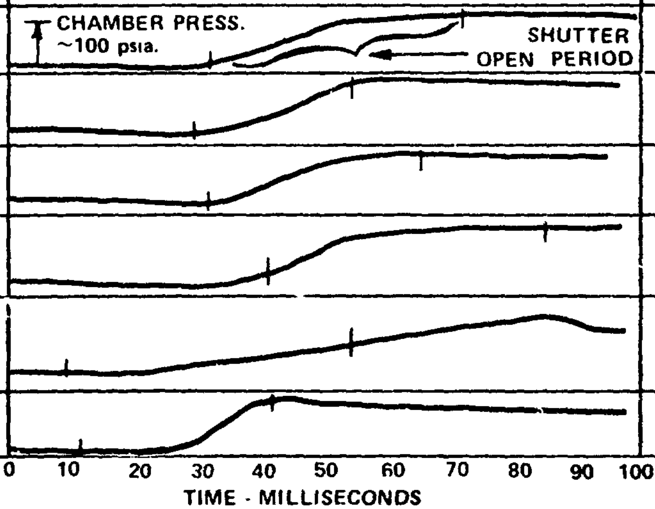
DATE	COLLECTOR NUMBER	BED TEMP.	CHAMBER PRESSURE & SHUTTER TIMING	COLLECTOR LOCATION, IN.		VISUAL OBSERVATION
				R	Z	
1/25	1	ROOM TEMP.		12	64	A FEW SNOW FLAKE SIGNATURES, SIGNATURES EVIDENT AND PARTICLES MARGINAL
1/25	2	ROOM TEMP.		14½	62	SIGNATURES EVIDENT PARTICLES MARGINAL
1/25	3	502°F		21	61	NO SIGNATURES AND NO PARTICLES
1/25	4	680°F		21	61	NO SIGNATURES AND NO PARTICLES
2/5	1	ROOM TEMP.		7¾	26¼	INTERNAL REFLECTION SPECTROSCOPY MEASUREMENT
2/5	3	ROOM TEMP.		2 7/16	20	HIGH DENSITY OF SIGNATURES SNOW FLAKE SIGNATURES SMALLER IN BOTH CASES
2/5	4	ROOM TEMP.		2-7/16	20	HIGH DENSITY OF LARGER SIGNATURES ~ 10 μm DIA. AND HIGH DENSITY OF DONUT SHAPES ~5 μm DIA.

0 10 20 30 40 50 60 70 80 90 100

TIME - MILLISECONDS

Table VI
FAIR II HYDRAZINE THRUSTER
POST-VIBRATION TEST DATA

DATE	COLLECTOR NUMBER	BED TEMP	CHAMBER PRESSURE & SHUTTER TIMING	COLLECTOR LOCATION R	COLLECTOR LOCATION Z	VISUAL OBSERVATION
2/20	1	ROOM TEMP.		10¾	29½	SOME SIGNATURES NO PARTICLES
2/20	2	ROOM TEMP.		10¾	26	NO SIGNATURES NO PARTICLES
2/20	3	ROOM TEMP.		10¾	24	NO SIGNATURES NO PARTICLES
2/20	4	ROOM TEMP.		10¾	24½	NO SIGNATURES NO PARTICLES
2/21	1	295°F		10¾	29½	VERY MARGINAL SIGNATURE INDICATIONS
2/21	2	605°F		10¾	26	VERY MARGINAL SIGNATURE INDICATIONS



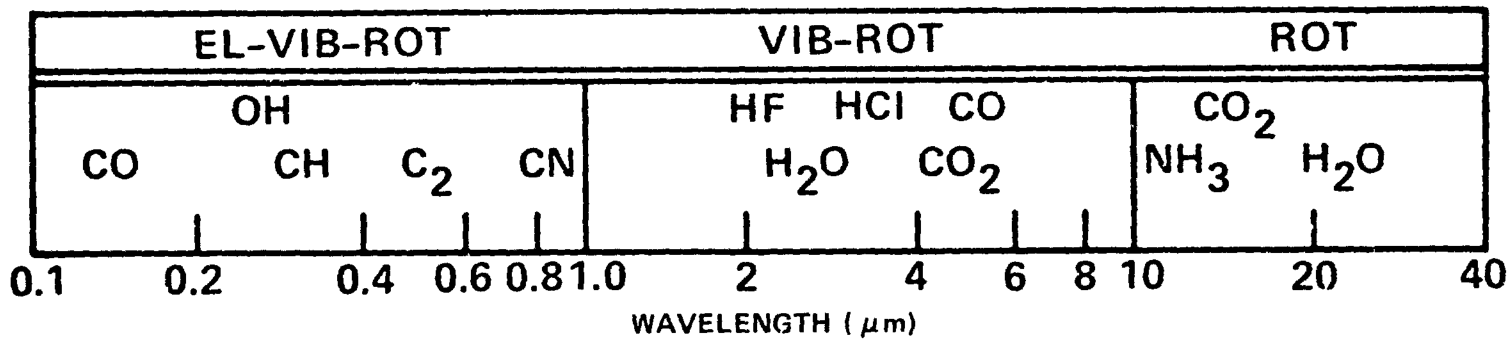


Figure 1 SPECTRAL RANGE OF IMPORTANT RADIATING SPECIES IN EXHAUST PLUMES

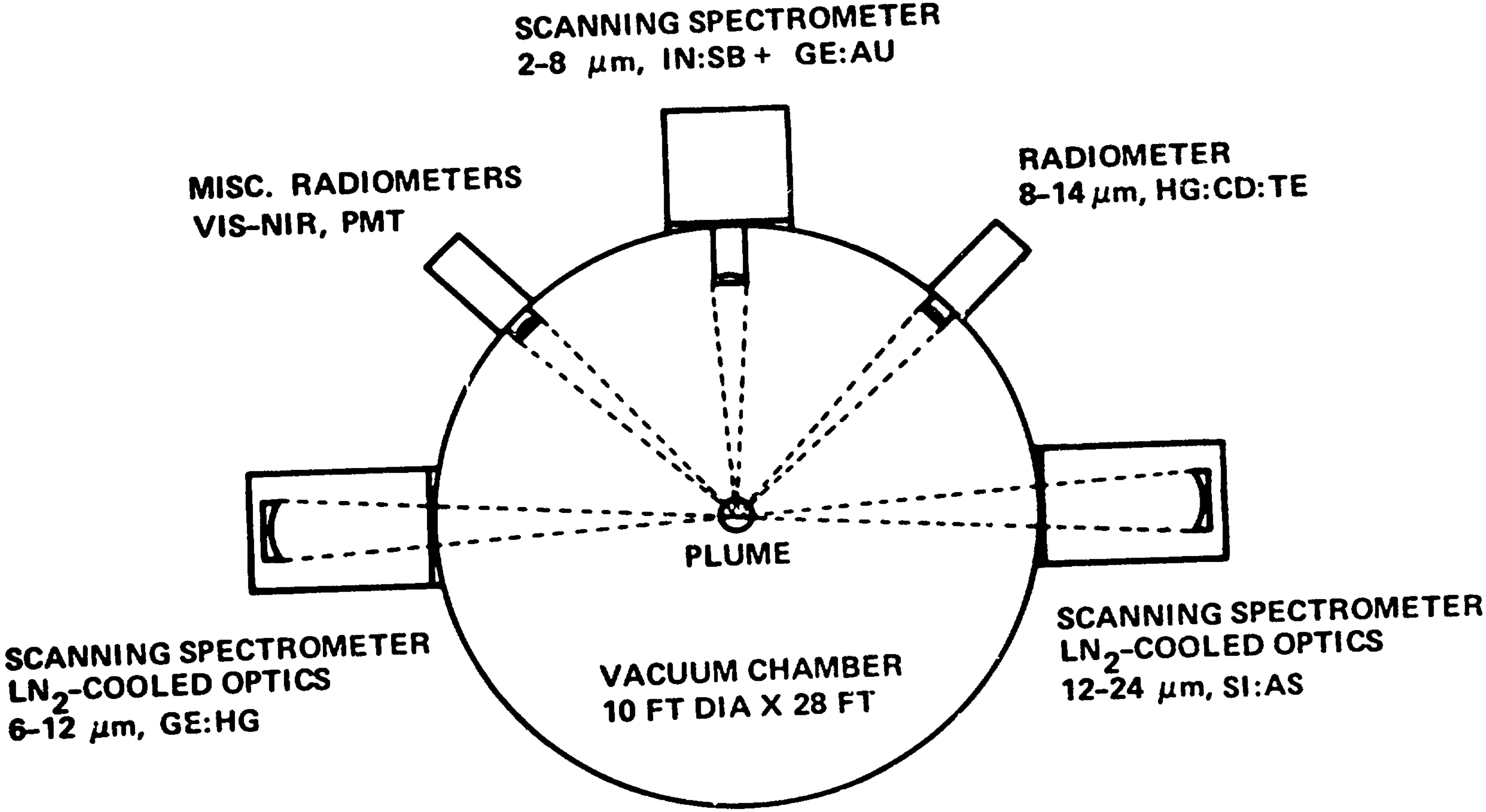


Figure 2 OPTICAL DIAGNOSTICS

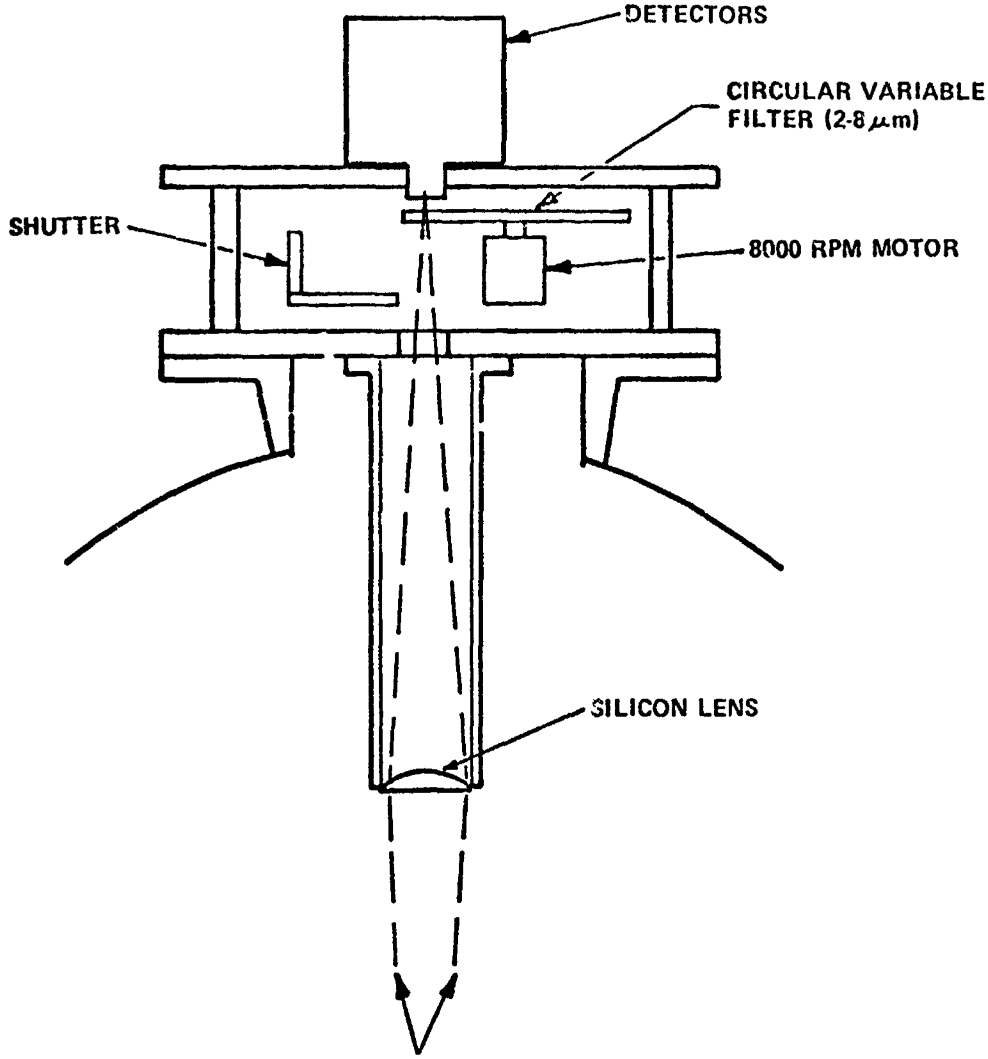


Figure 3 SCHEMATIC OF SWIR RAPID-SCAN SPECTROMETER

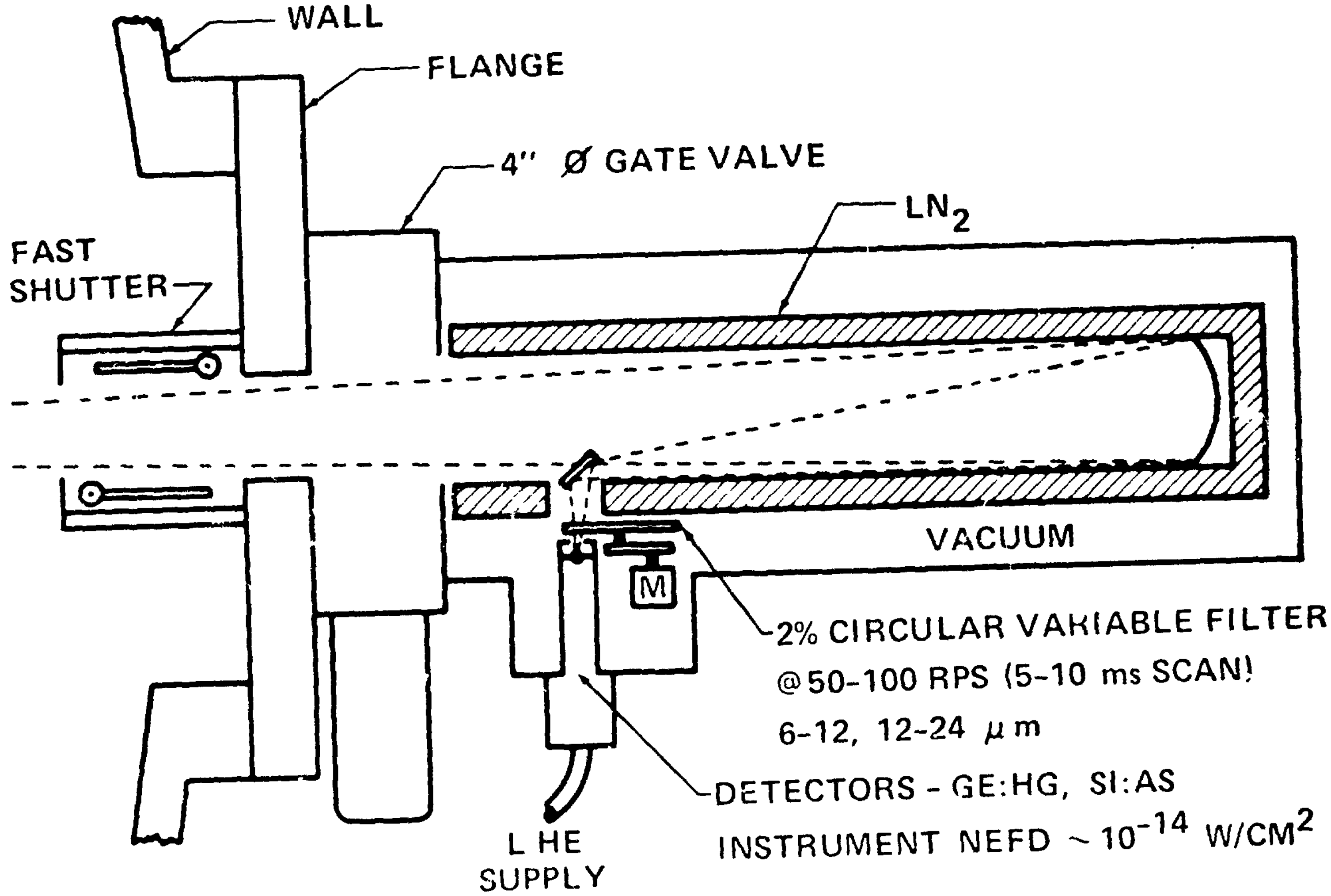


Figure 4a FAR IR SCANNING SPECTROMETERS

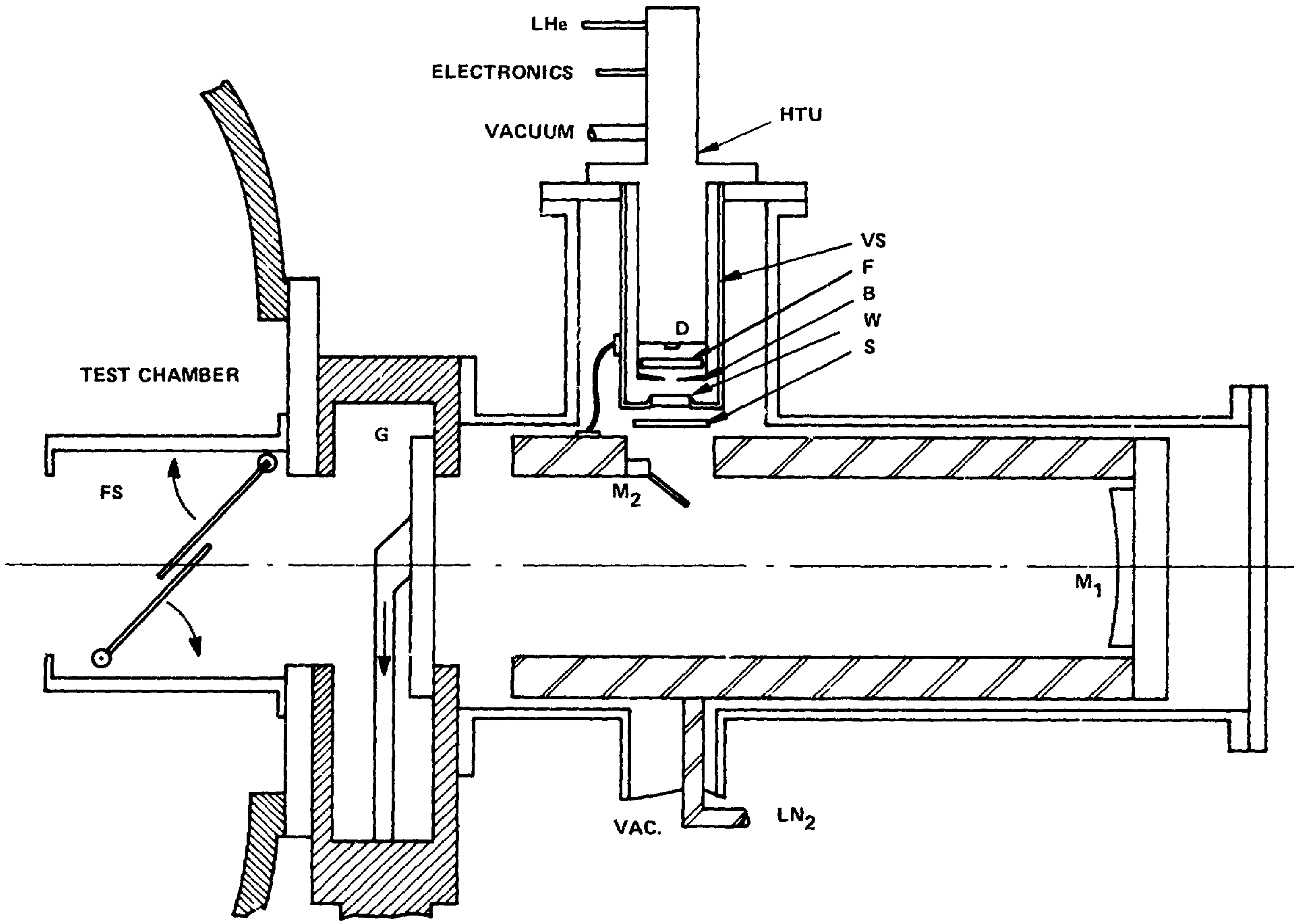
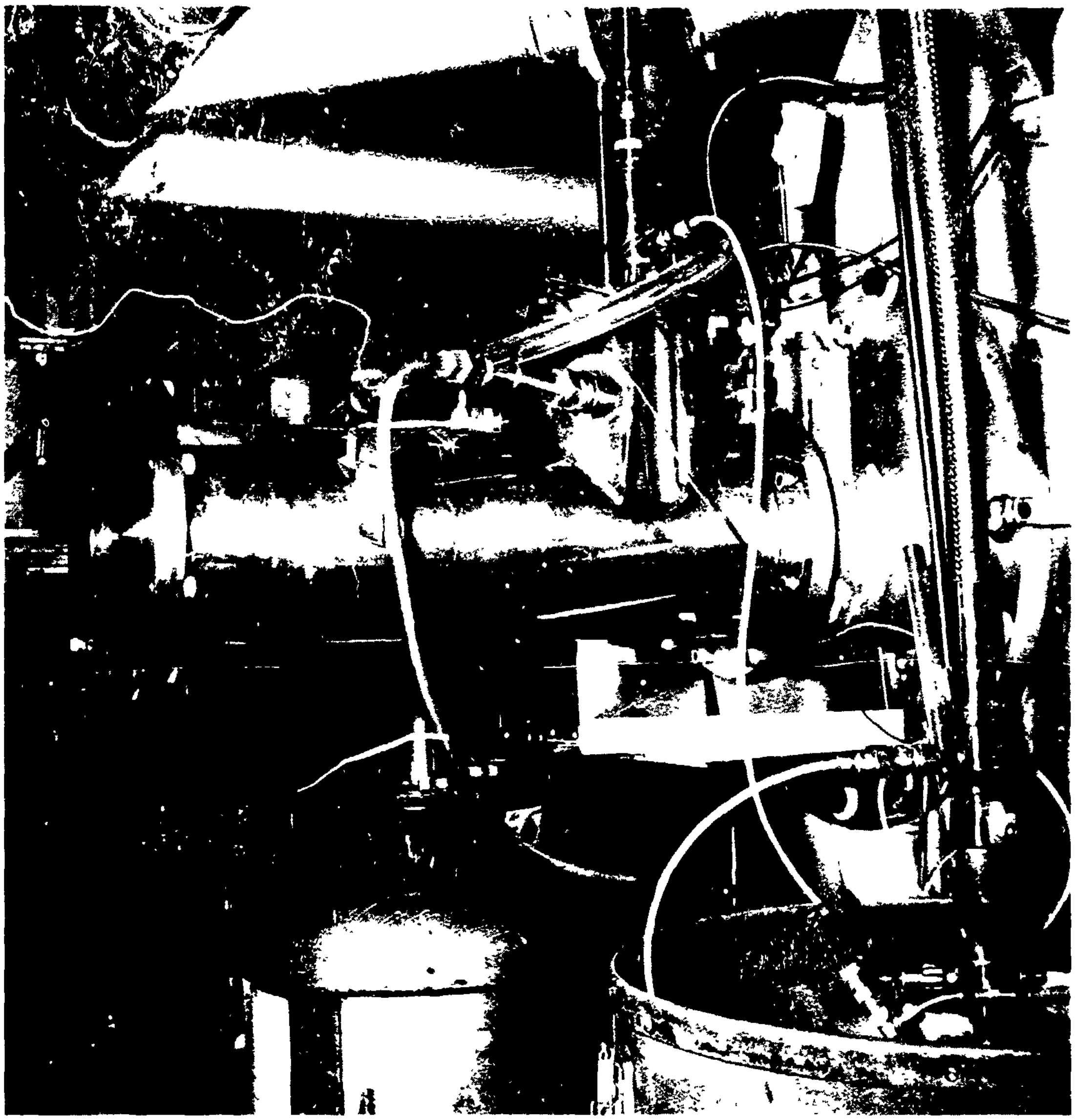
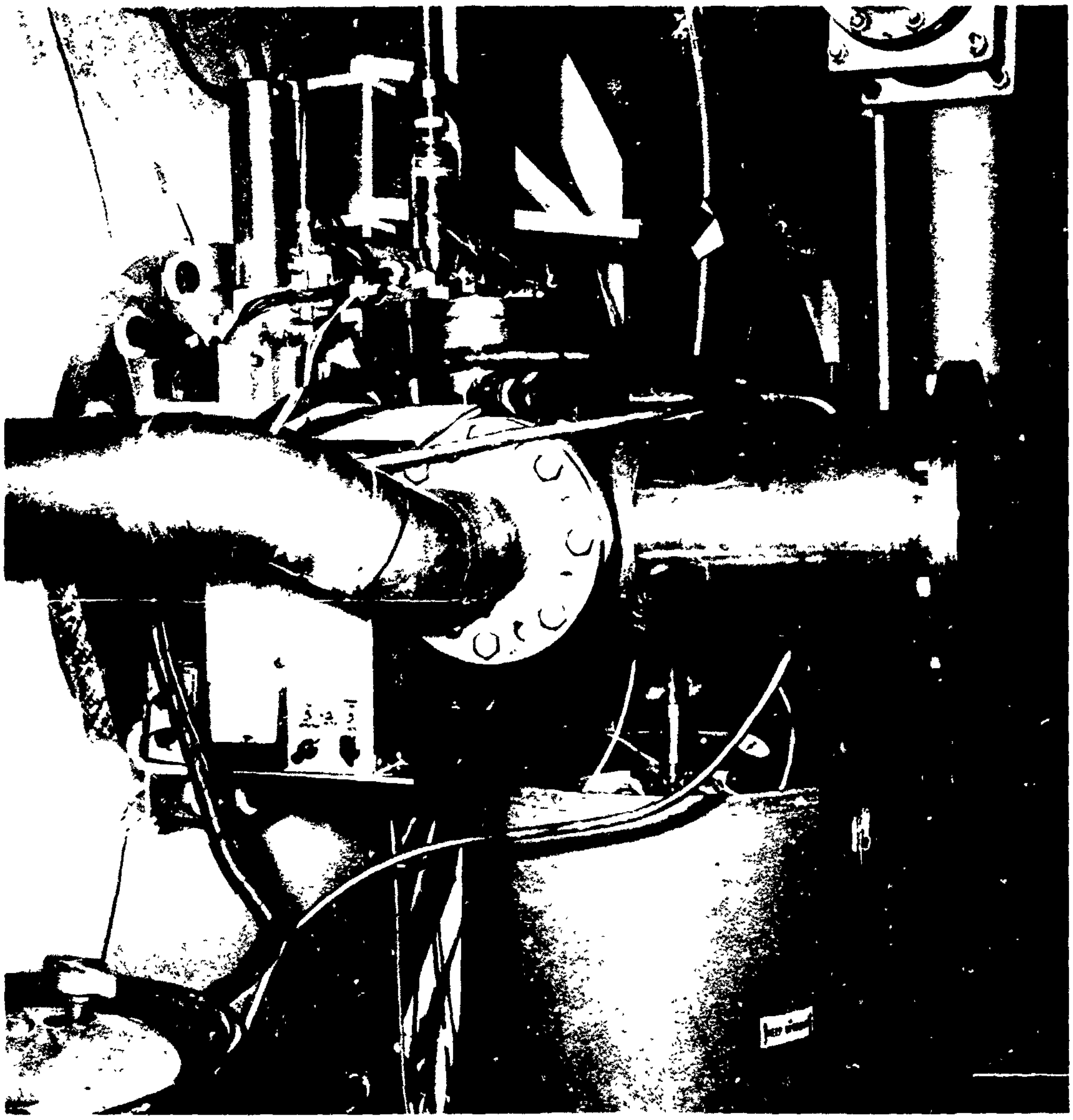


Figure 4b SCHEMATIC DIAGRAM OF ONE SPECTROMETER



(a) ASSEMBLY ON HIGH-ALTITUDE TEST CHAMBER

Figure 5 LWIR SPECTROMETER ASSEMBLY



(b) VIEW OF ASSEMBLY SHOWING LARGE VACUUM LINE

Figure 5 LWIR SPECTROMETER ASSEMBLY

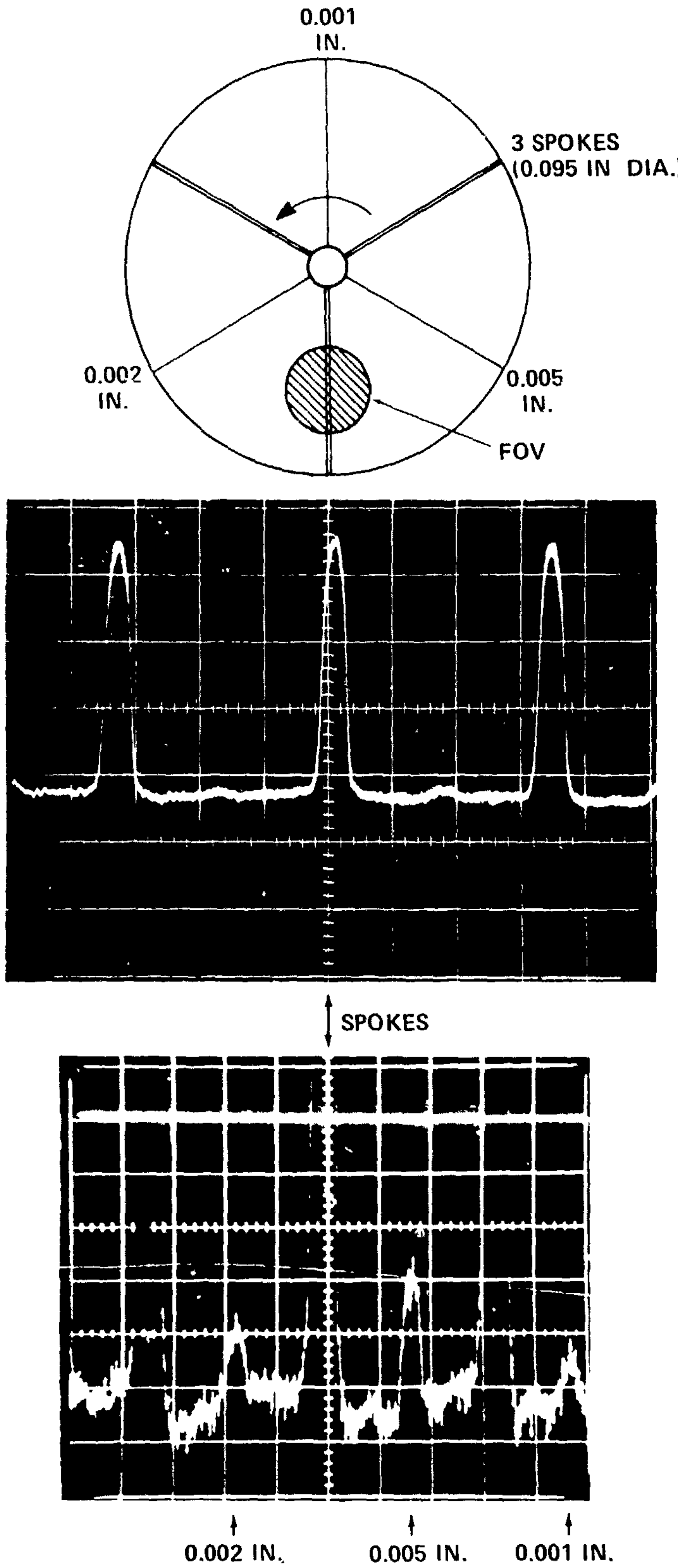


Figure 6 RADIOMETER CALIBRATION SYSTEM

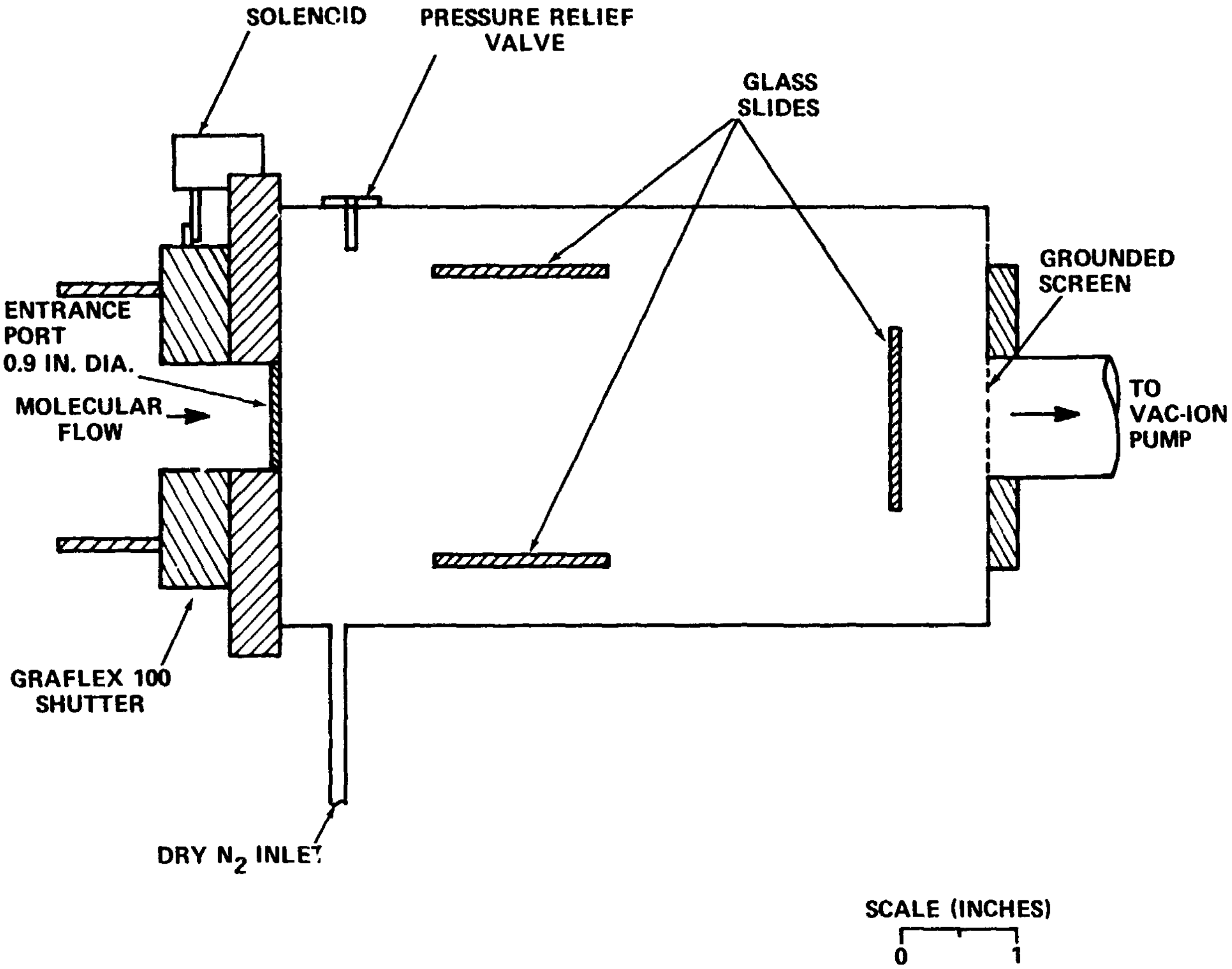


Figure 7 SCHEMATIC OF PARTICLE COLLECTOR

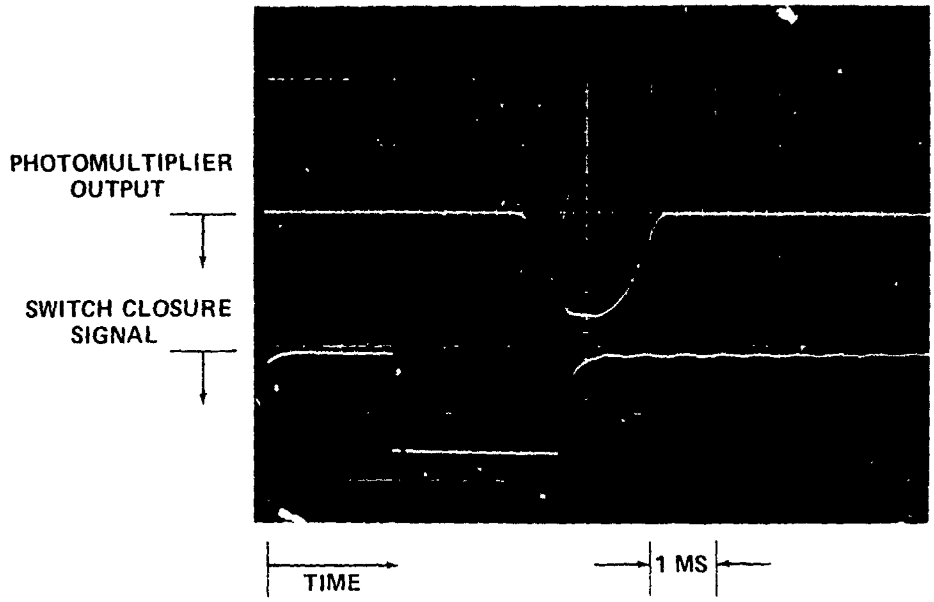


Figure 8 TYPICAL PARTICLE COLLECTOR CALIBRATION

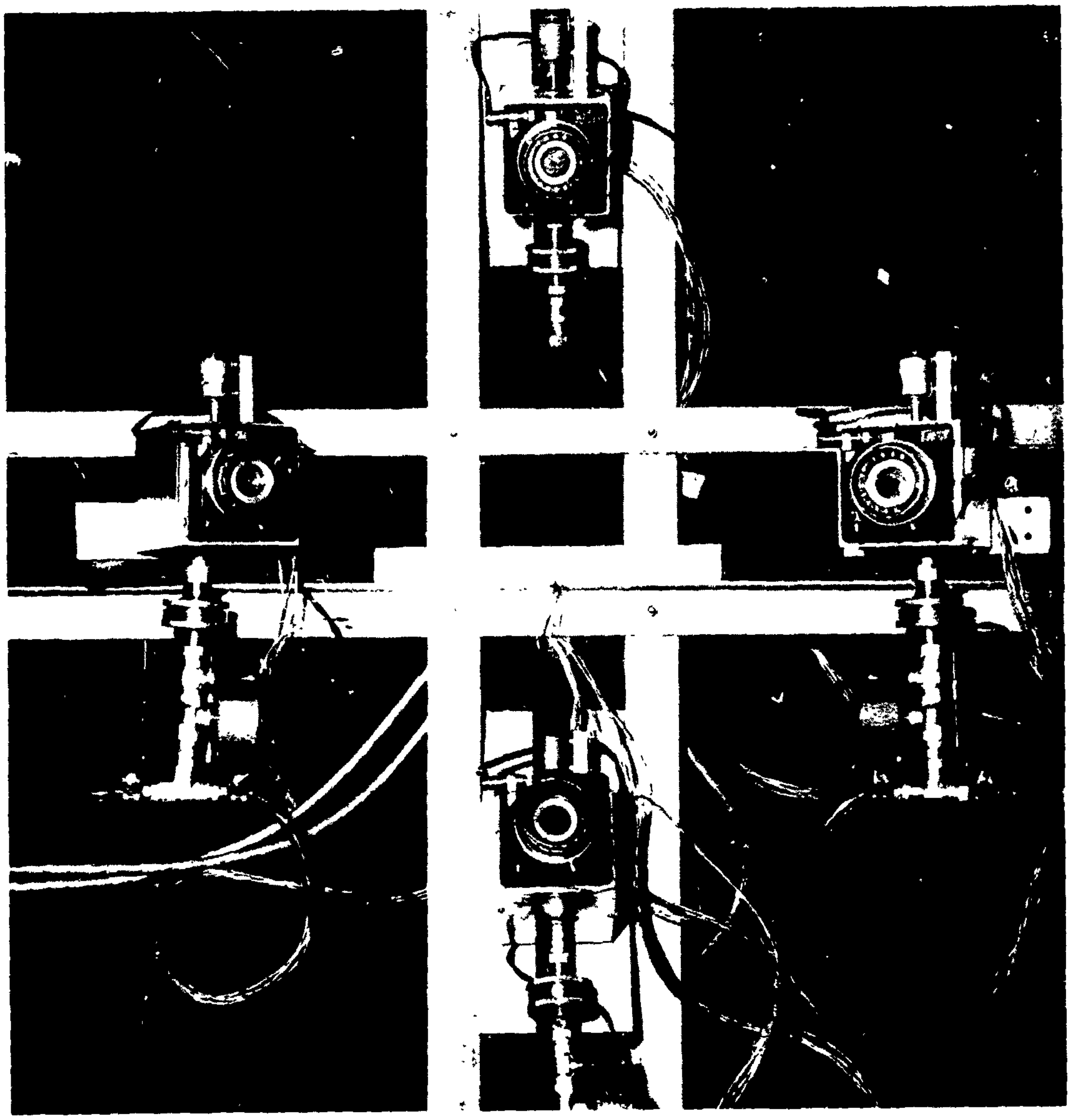


Figure 9 INSTALLATION OF FOUR PARTICLE COLLECTOR ASSEMBLIES IN HIGH-ALTITUDE TEST FACILITY

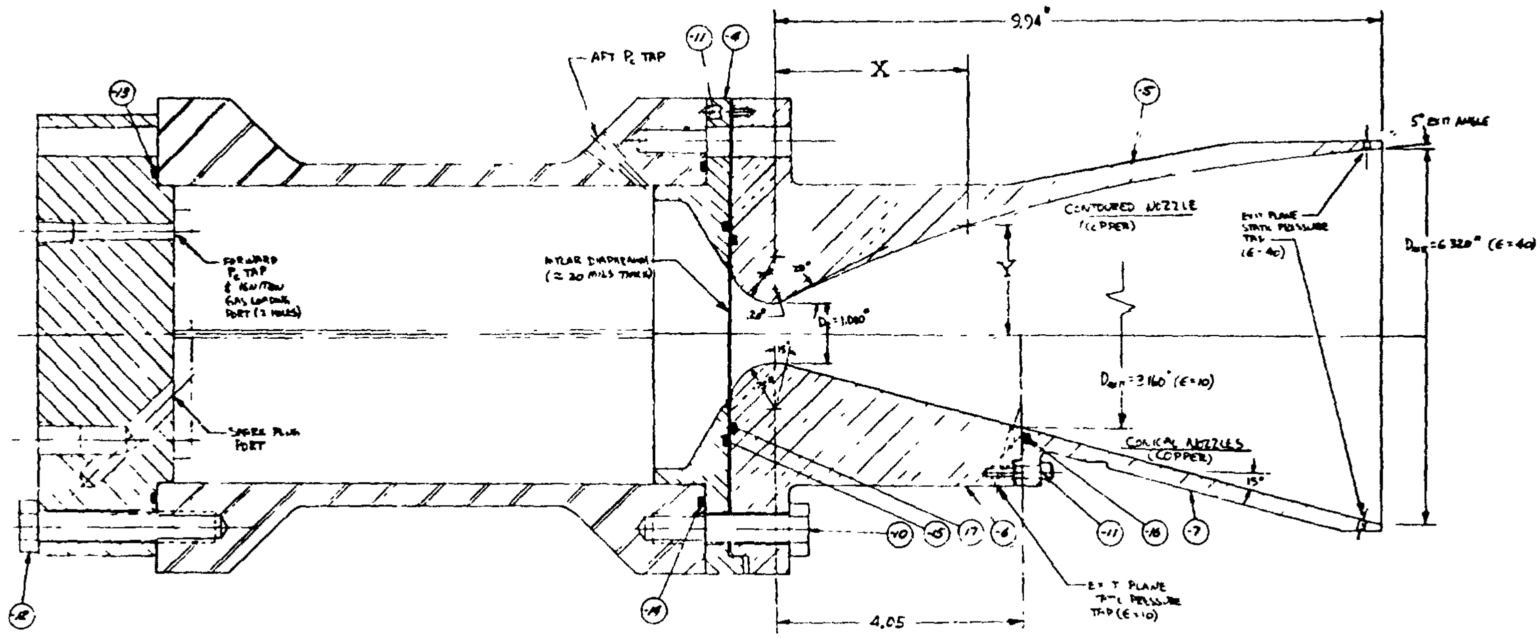


Figure 10 SOLID PROPELLANT RESEARCH MOTOR

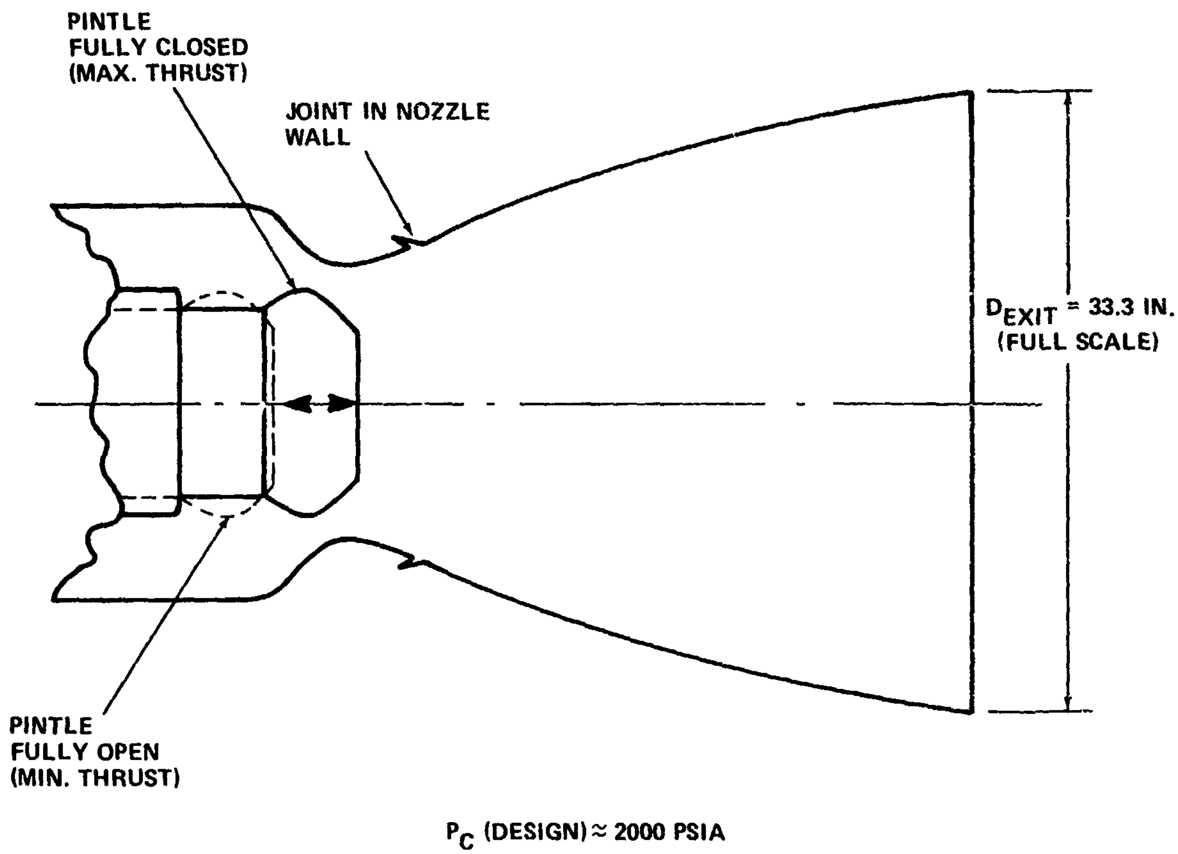


Figure 11 PINTLE NOZZLE CONFIGURATION

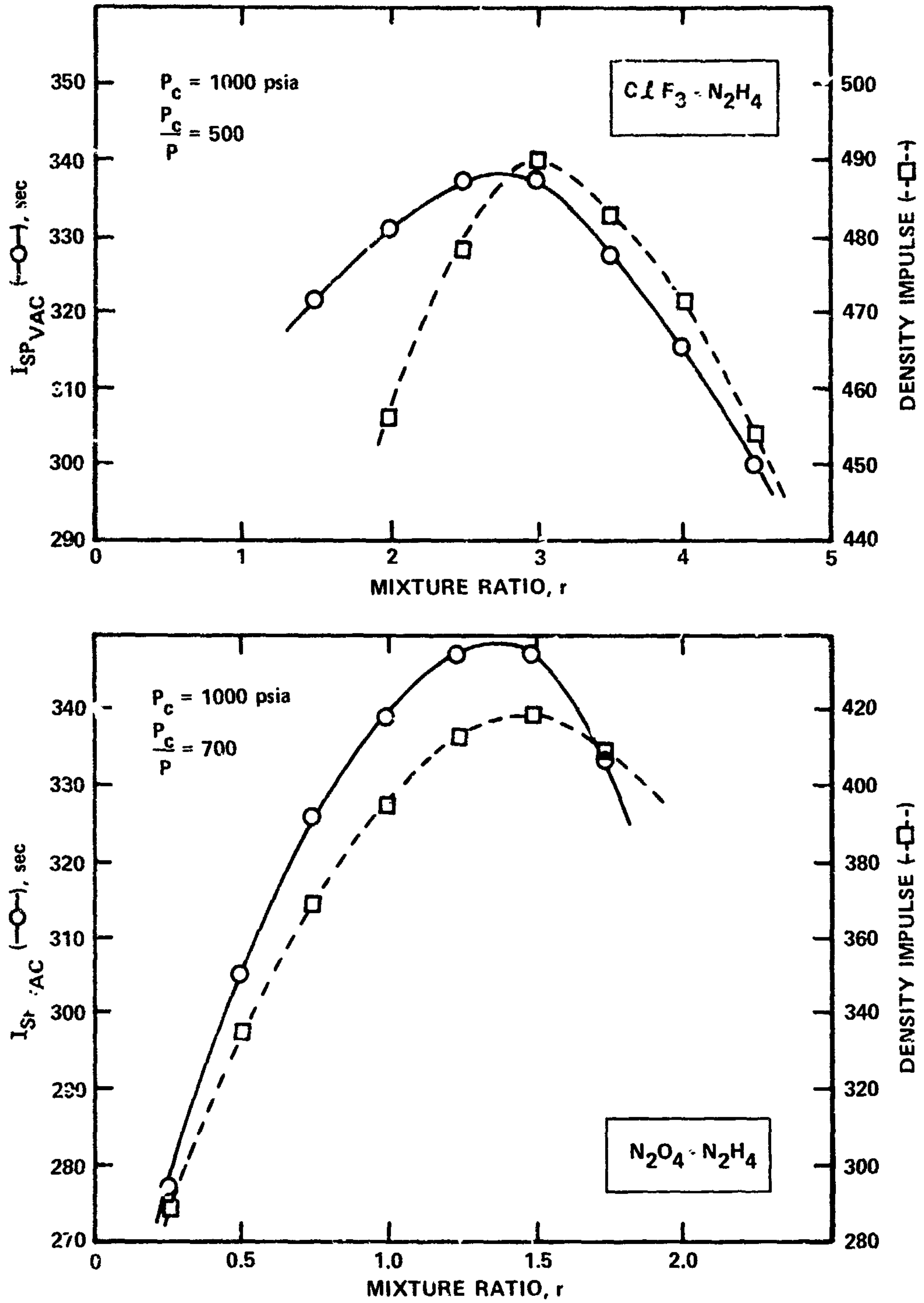


Figure 12 CALCULATED PERFORMANCE OF TWO PROPELLANT COMBINATIONS AS FUNCTION OF MIXTURE RATIO

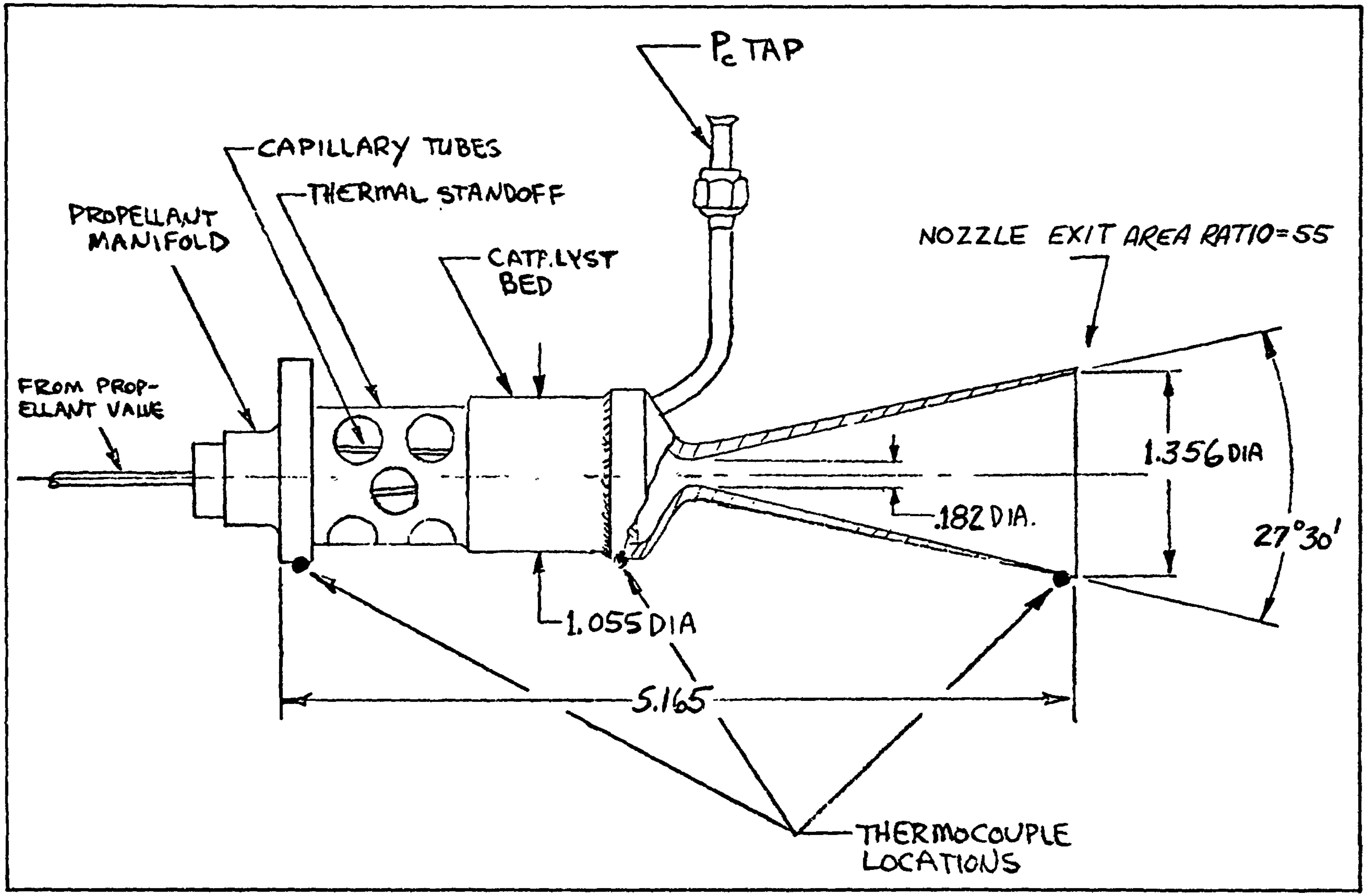


Figure 13 HYDRAZINE THRUSTER

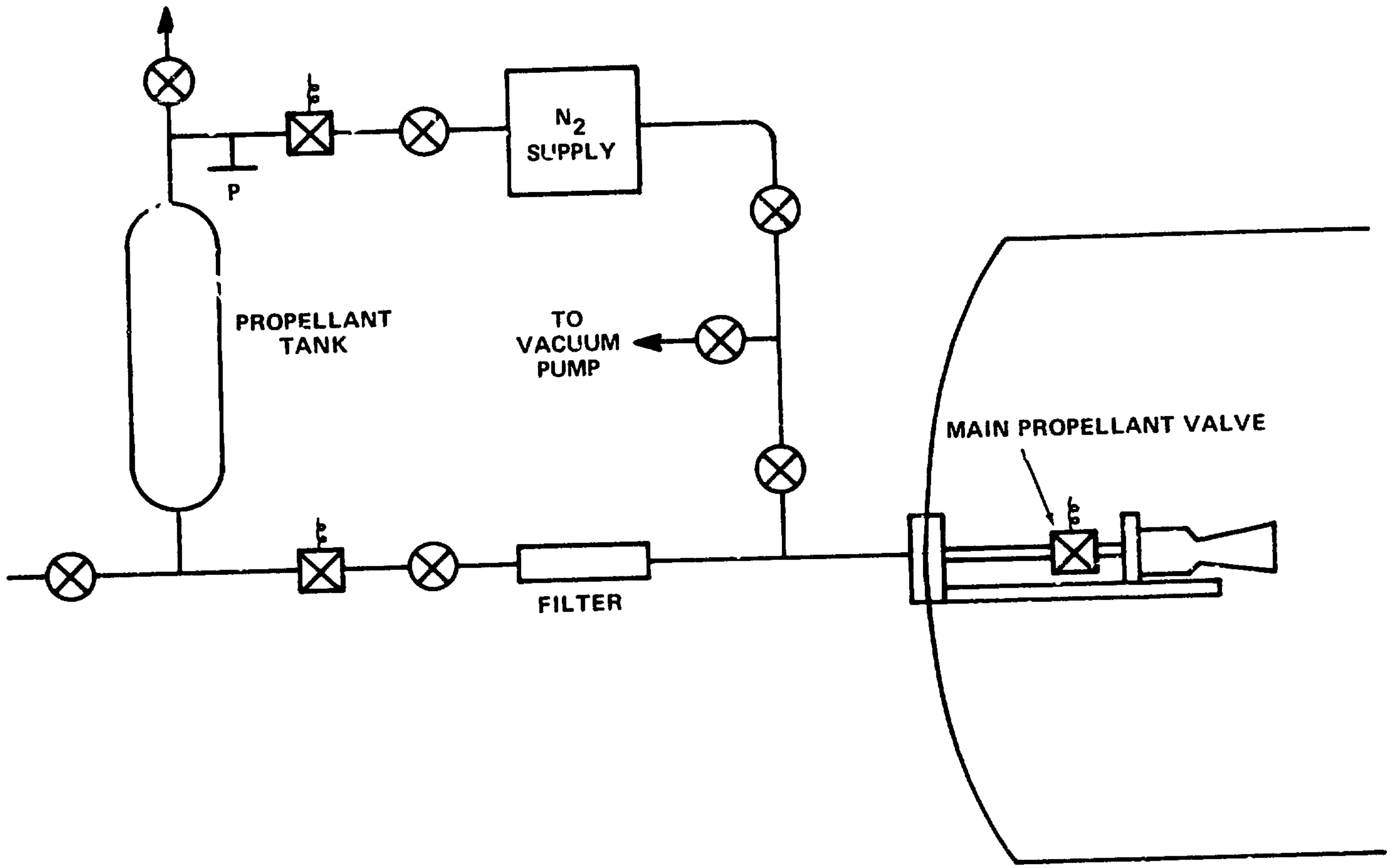


Figure 14 SCHEMATIC OF PROPELLANT SYSTEM FOR HYDRAZINE THRUSTER

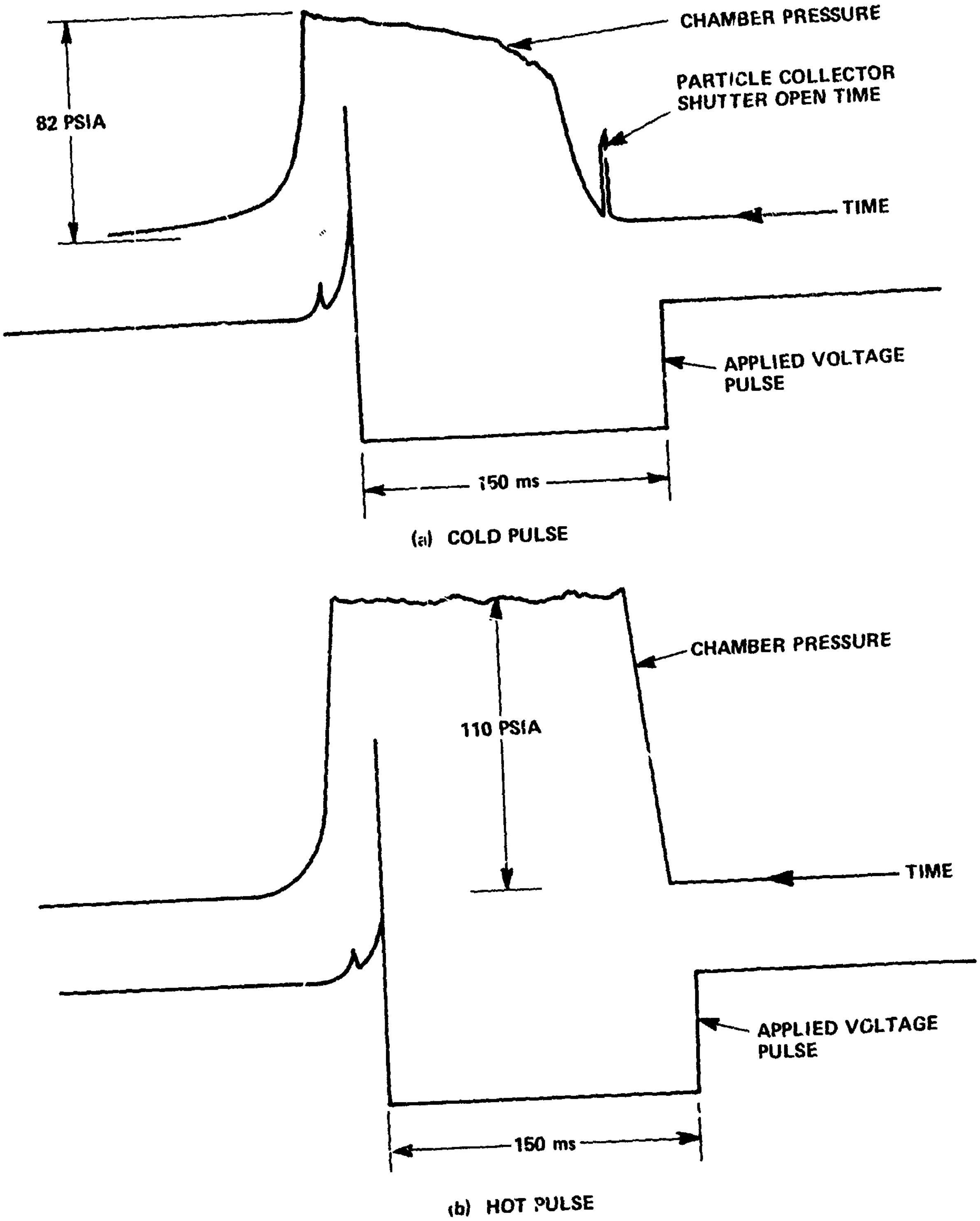


Figure 15 TYPICAL PULSE SEQUENCE OF EVENTS

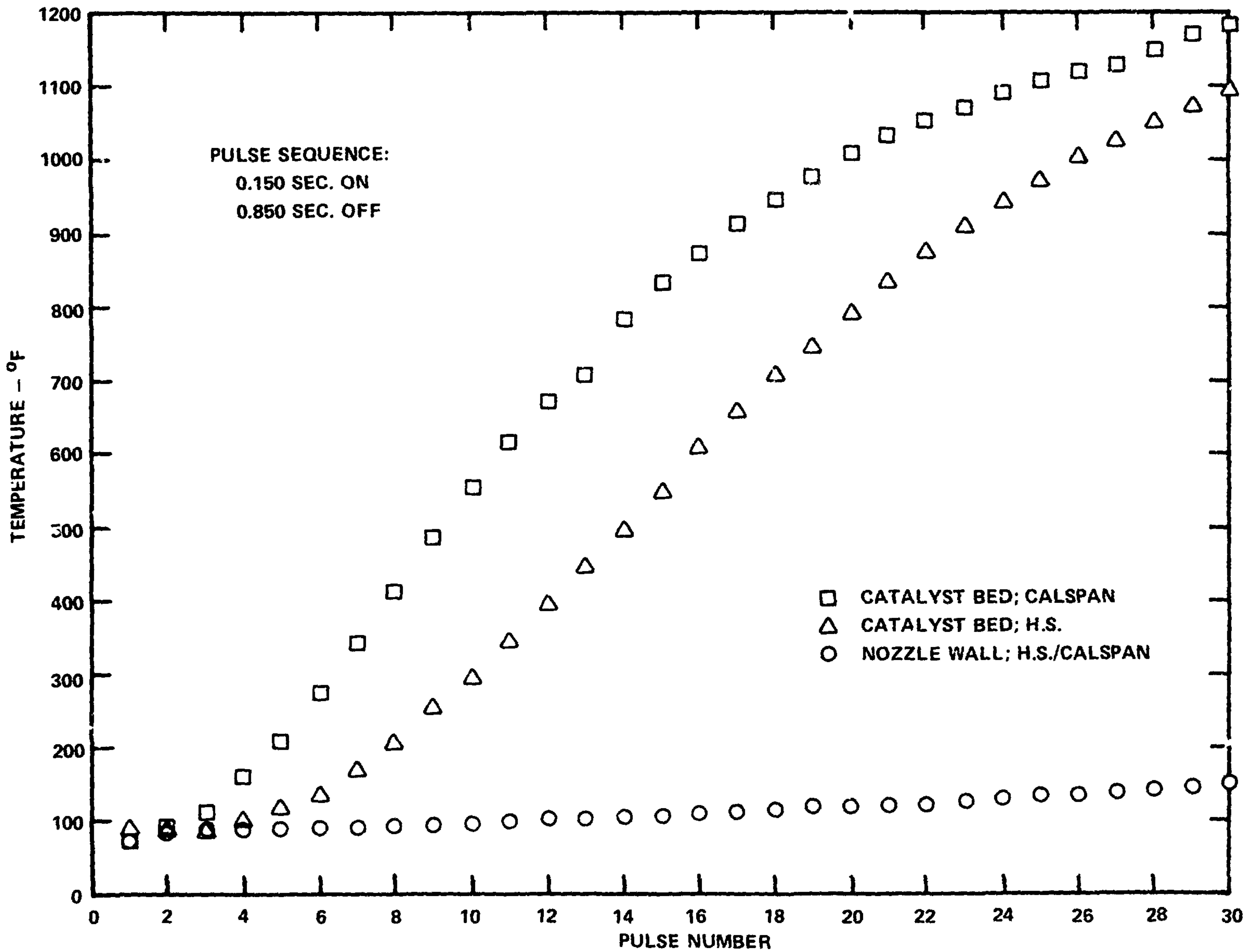


Figure 16 N_2H_4 THRUSTER TEMPERATURE HISTORY: 30-PULSE SEQUENCE

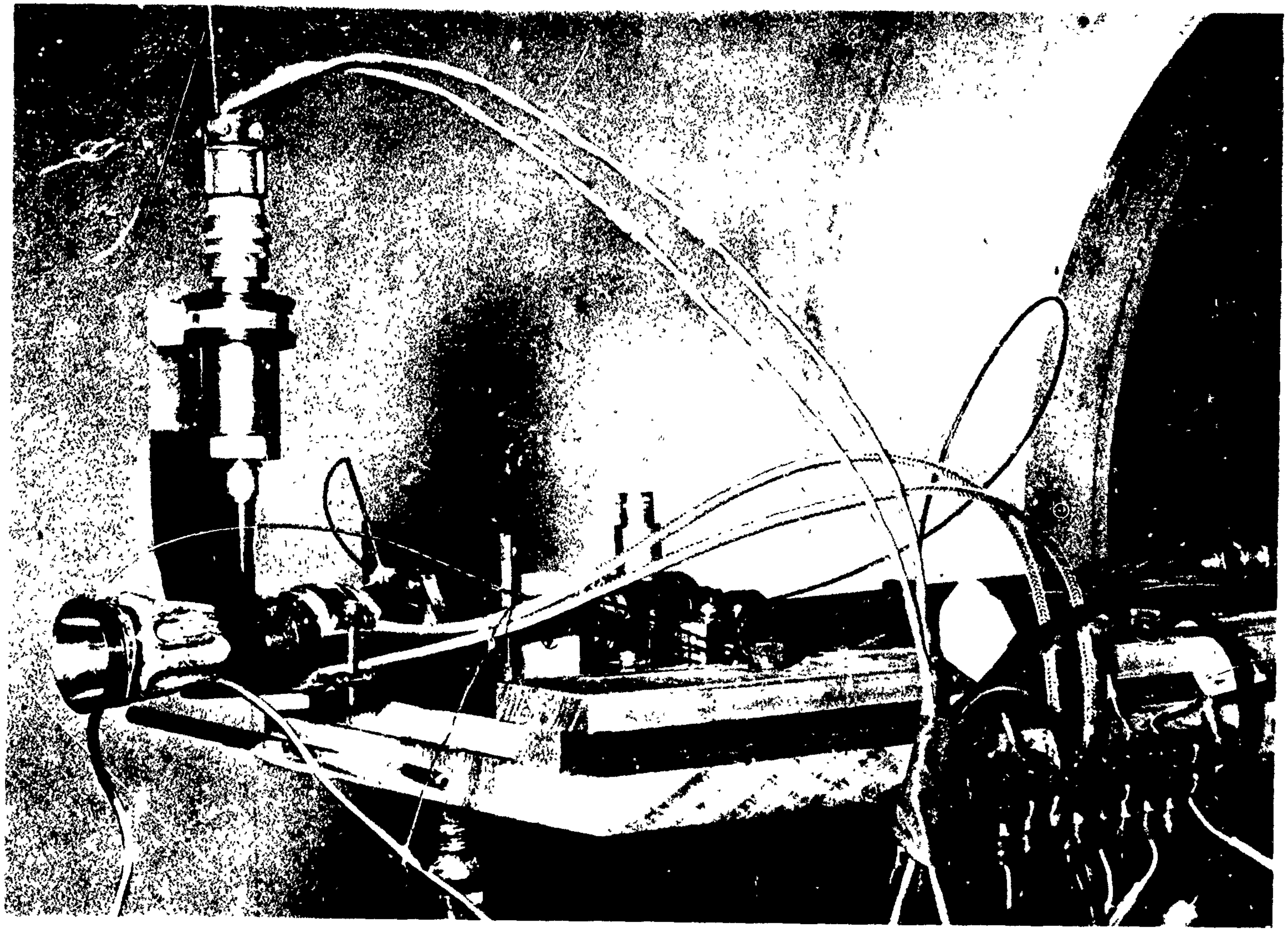
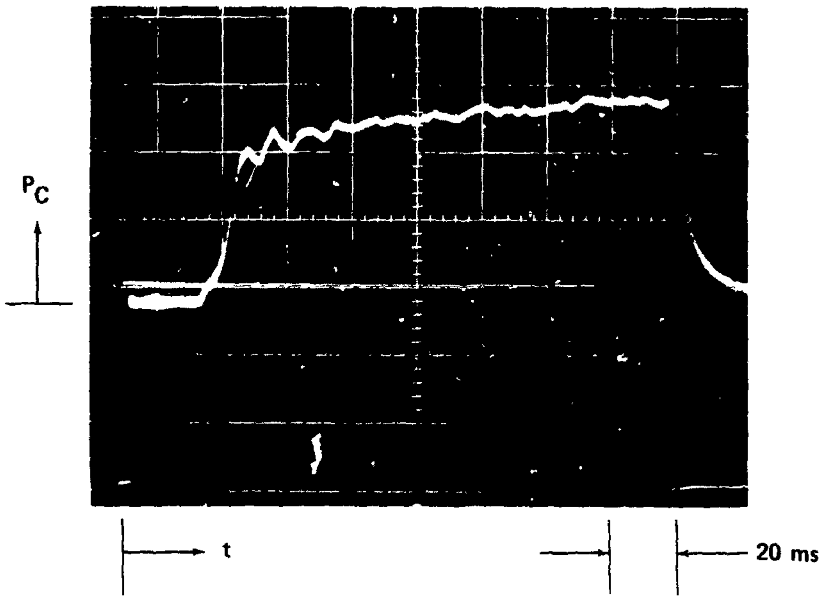
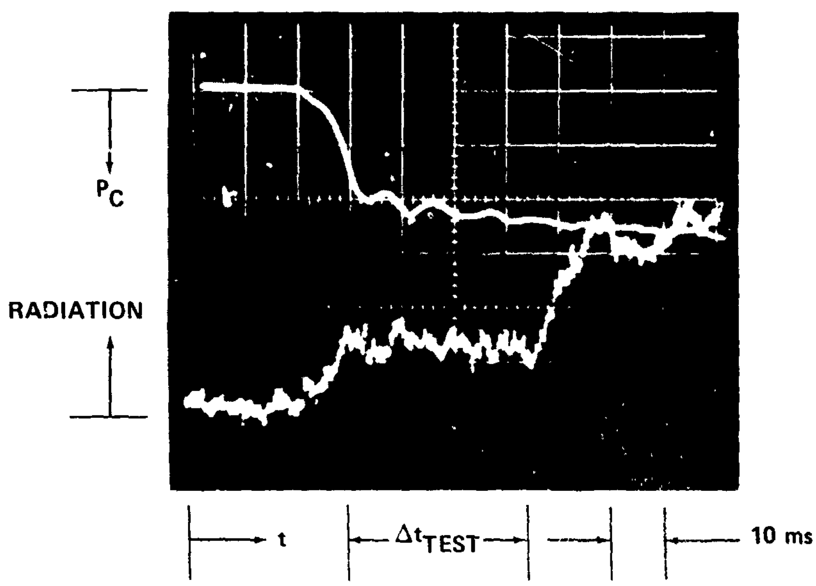


Figure 17 HYDRAZINE THRUSTER INSTALLED IN HIGH-ALTITUDE TEST FACILITY

HYDRAZINE THRUSTER PULSE
$T_{BED} \approx 300^{o}F$ (HEATER)



(a) CHAMBER PRESSURE



(b) CHAMBER PRESSURE AND RADIATION MEASUREMENT

Figure 18 SINGLE-PULSE PRESSURE AND RADIATION DATA

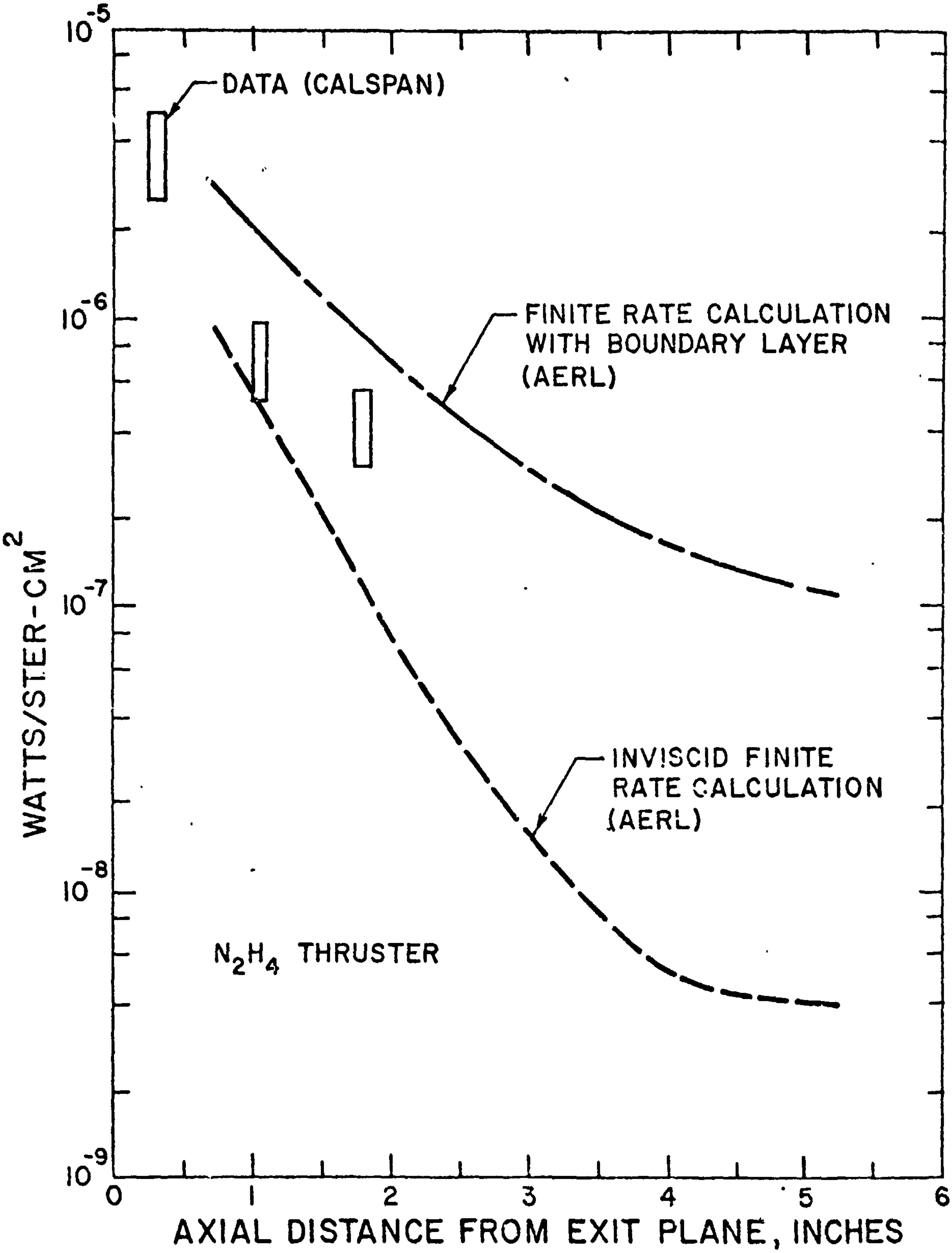
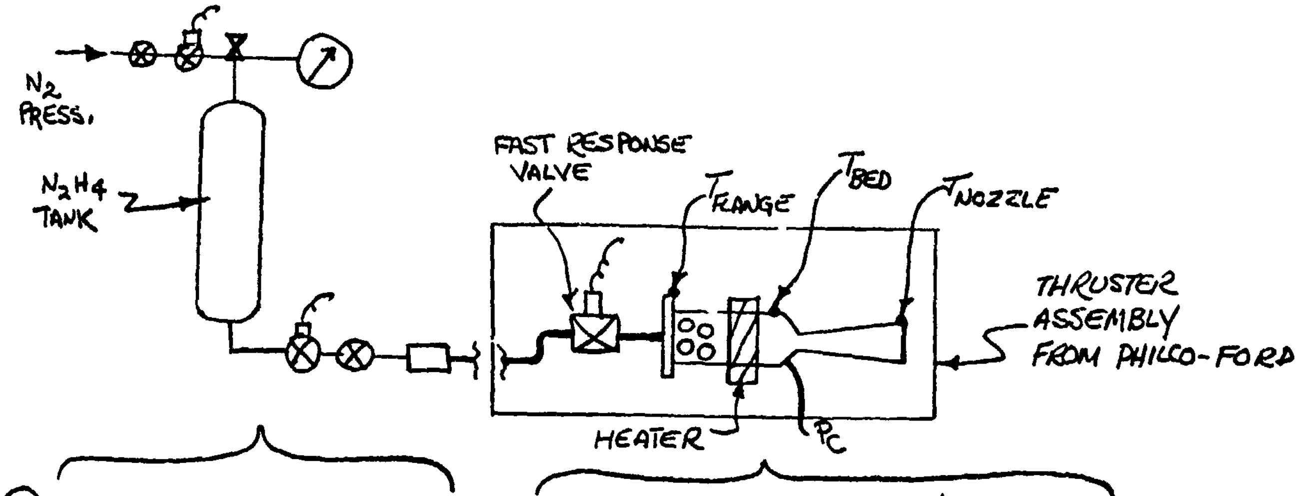


Figure 19 NH_3 IN-BAND RADIANCE (7.97-12.2 μm) VS. DOWNSTREAM DISTANCE; HYDRAZINE MOTOR

APPENDIX A

HAMILTON STANDARD WORK STATEMENT FOR HYDRAZINE THRUSTER MODIFICATION

The major items are shown schema cally as follows:



(A) PROPELLANT SUPPLY SYSTEM

- DESIGN, FABRICATE ASSEMBLE, CHECKOUT A PRESSURIZED N_2H_4 SYSTEM
- PRECHARGE WITH N_2H_4
- DELIVER, ACTIVATE SYSTEM, PROVIDE TECHNICAL ASSISTANCE AT CALSPAN

(B) ENGINE PREPARATION & TESTING

- PULSE THRUSTER/FAIR II DUTY CYCLE: DETERMINE HDWE. TEMPS. AS FN. OF TIME. COLLECT & ANALYZE EXHAUST SAMPLE
- INSTALL HEATER AROUND CATALYST BED & DUPLICATE SAME TEMPERATURE DISTRIBUTIONS
- SINGLE PULSE FIRE ARTIFICIALLY HEATED THRUSTER. COLLECT & ANALYZE EXHAUST SAMPLE
- REMOVE HEATER; VIBRATE THRUSTER ASSY. PER FAIR II ACCEPTANCE CRITERION
- REINSTALL HEATER/DELIVER

Work Statement

Engine Preparation and Testing

It is desired to simulate the operation of a recently fired hydrazine engine in space. Facility limitations at Calspan preclude firing the engine to warm it to operating conditions. An external source of heat (electrical) must therefore be used to heat the engine.

One REA 16-5 thruster will be provided to Hamilton Standard by Calspan. H-S will test the engine in "as is" condition to establish the temperature profile and temperature-time histories at the:

Engine mounting flange

Nozzle boss

Nozzle exit zone

These temperatures will be measured with chromel/alumel thermocouples welded to the engine. These thermocouples will be delivered to Calspan with the engine for further thermal profile correlation in their facility.

The temperature profile will be determined at the conclusion of a train of 30 pulses of 0.150 sec duration with an "off" time of 0.850 sec between pulses. A heater will then be attached to the bed area of the engine and the bed artificially heated to the temperature measured at the conclusion of the above firing duty cycle. If the resulting nozzle temperature is more than 100°F lower than that recorded at the conclusion of the firing duty cycle, an additional heater will be attached to the nozzle. Heater power level(s) will be noted to assist Calspan simulation.

Gaseous products will be collected from the engine during the last pulse of the 30-pulse firing duty cycle, and the product analyzed for:

NH_3, N_2, H_2.

With the engine artificially heated by the resistance heater(s), a single pulse will be fired. The resulting products will be collected and similarly analyzed for comparison.

Heater location will be noted, the heater(s) removed, and the REA vibrated according to the same random vibration acceptance criteria as specified by Philco-Ford for FAIR II thruster hardware. The engine will be then packaged for shipment.

Delivery of the vibrated engine with (3) thermocouples, heater(s) as required, gas analysis of two samples, and analog pulse data of both firings will constitute completion of this phase of the program.

Engine Propellant Supply System

To support the test series of the Hamilton Standard REA 16-5 at Calspan, a system capable of supplying hydrazine to the engine is required. This system shall have the following characteristics:

Propellant storage	2 lb max*
Flow	$\approx$ 0.025 lb/sec
Pressure	300 psia max
Filtration	5 μm nom.
Pressurizing and venting	Manual & elec. **
Arming & dumping	Manual & elec. **
Pressure relief	350 psia
Pressure readout (P_{tank} & $P_{chamber}$)	0-300 psia ***

* H-S to precharge with 1 lb N_2H_4

** Calspan to provide wiring & switching from connectors on valves

*** Calspan to provide wiring and readout

All electrical components to be provided with mating connectors. Component mounting structure and engine mounting will be provided by Calspan.

Hamilton Standard will provide the following effort to meet the above requirement:

- Design
- Procure or fabricate components
- Trial assembly
- Teardown and LOX cleaning
- Assembly
- Proof, leak and functional checkout
- Quality Control inspection
- Charge with N_2H_4
- Blanket with GN_2 (hermetic)
- Package for shipment

Delivery of the engine test system to Calspan constitutes completion of this phase of the program.

Comments

Because of shipping regulations, actual pre-filling of the propellant tank with hydrazine was not performed as planned at H-S. Rather, the filling was performed at Calspan by an H-S test engineer.

In the engine preparation and testing portion of the effort, the 30-pulse firing sequence was performed at H-S without incident and an exhaust gas sample was obtained on the last pulse. Analysis of this sample showed 33% H_2, 26% N_2, and 41% NH_3 (this analysis corresponds to $\approx$ 35% NH_3 dissociation). Final catalyst bed temperature was measured to be $\approx$ 1100°F. Electrical heating of the catalyst bed was next attempted, employing a wrap-around, 80-watt heater. After initial problems involving excessive heat losses at the mounting flange were resolved, the catalyst bed was successfully heated to 700°F at which time the heater failed. Although this bed temperature was considerably below the desired 1100°F, a single-pulse gas sample was obtained for comparison with the previous hot-firing pulse sample. This sample showed only 15% NH_3 dissociation (compared to 35% at 1100°F), confirming the need for a higher bed temperature. Additional heaters with a higher temperature rating and a better fit to the motor catalyst bed were then procured by H-S. Heating of the bed to temperatures up to 1050°F was successfully demonstrated and a single-pulse firing was performed with the artificially heated bed.

All control wiring, pulse sequences, and other related support equipment were developed at Calspan. Upon completion of preliminary firings in the Calspan altitude chamber to check out the performance of the motor and propellant supply system, the thruster assembly was returned to H-S for the vibration sequence. Upon return of the motor to Calspan the post-vibration pulse sequence firings were undertaken.